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FOREWORD

The Quarterly Reliability Status Report is submitted in accordance with the Apollo documentation requirements delineated in NASA Contract NAS9-150, Paragraph 4.5.4 7, of "Project Apollo Spacecraft Development Statement of Work," Part 4, dated 18 December 1961, and MIL-R-27542, Paragraph 5.4.3. The information contained herein covers the period from 1 January through 31 March 1963.

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I. RELIABILITY PROGRAM IMPLEMENTATION

STATUS OF PROGRAM PLAN

The Apollo Reliability Program Plan was reissued to include program redirection, the LOR mission concepts, and appropriate changes resulting from NASA comments. The following inclusions to the plan were made.

1. Organization and Milestones. Current organizational charts indicate the latest changes in NAA/S&ID structure and personnel status. A chart was added that delineates the functions and responsibilities of the Apollo Engineering organization. Reliability milestone charts illustrating completed and scheduled milestones were extended through 1963.

2. Apportionments. LOR mission concepts are reflected in the reapportionments for both the Apollo/Saturn system and the spacecraft systems.

3. Preferred Parts Manual. The Preferred Parts List is now identified as the Reliability Preferred Parts Manual. The manual was published and is now in use for the Apollo system.

4. Method of Mechanical Analyses. A description of the failure effects analysis is included describing the method of analysis that is employed in all spacecraft systems and components and on GSE mission essential equipment.

5. Reliability/Crew Safety Design Review. The design review schedule was expanded to cover activities into July 1963, as well as noting those reviews which have been accomplished.

6. Data Management Organization. Further clarification of the functions and activities of the Data Management organization was provided, including the use of the Data Center for information flow for machine program analysis techniques.

7. Reliability Assessment. Reliability assessment methods based on the redefinition of the qualification test philosophy and plan, as defined in Reoriented Apollo Qualification Test Program

(SID 62-1405), are included. Reliability demonstration concepts have been replaced with assessment reliability concepts.

8. Nonconformance Analysis. Reliability engineering responsibilities in performing failure analysis were further clarified. Sample nonconformance and nonconformance analysis reports were inserted for illustration.

Implementation of the inclusions to the program plan is presently in effect. Revisions will be accomplished, as necessary, to provide a comprehensive view of the status of reliability activity on the Apollo Program.

STATUS OF GROUND QUALIFICATION TEST PLAN

INTRODUCTION

By mutual agreement reached at meetings between NASA and S&ID during the past quarter, the Reliability-Qualification Test Plan (SID 62-204) has been set aside as the document delineating the qualification testing for the subsystems comprising the Apollo spacecraft system. A component and system qualification test plan has been prepared to reflect the reoriented concepts and program implementation developed during those meetings. This plan will be known as the Ground Qualification Tests and proposes to replace the Reliability-Qualification Test Plan. It has been incorporated into the Apollo General Test Plan as Volume III (SID 62-109-3), issued in March 1963.

The program has been planned for individual systems and associated ground support equipment to achieve ground qualification prior to first manned flight with minimum hardware requirements. This has been accomplished by clearly defining the scope of each test program, delineating the tests to be performed with an optimized utilization of hardware, and establishing minimum test criteria consistent with equipment criticality and mission requirements.

The test plan has been designed to conform to the requirements of the Apollo Spacecraft Development Statement of Work, dated 18 December 1961, and has been guided by the provisions of NASA quality publication, Quality Assurance Provisions for Space System Contractors (NCP 200-2), dated 15 December 1961, and MIL-R-27542 (USAF), dated 28 June 1961, Amendment No. 1, dated 31 October 1961.

SCOPE

The objective of the Ground Qualification Tests is to delineate and implement the philosophy to be employed in verifying the design integrity for the Apollo spacecraft systems and associated ground support equipment to accomplish the intended mission.

Ground qualification will be achieved by demonstration of system capability to perform within specification requirements at three levels of system testing: component, system, and integrated systems. Ground qualification will demonstrate that functional performance and quality

provisions of all systems and associated GSE are repeatable on randomly selected test articles.

PROGRAM LOGIC

Unique Requirement

Since the time phasing of the Apollo Project prohibits a statistically rigorous demonstration of all essential design parameters, a sound engineering approach toward insuring adequate system reliability was applied as an optimum compromise. This can be achieved by designing development tests that seek out the potential causes of failure and by applying qualification tests that verify that the causes of failure have been eliminated. The qualification test plan, therefore, includes a logical combination of tests designed to verify the elimination of all potential weak points or failure modes and demonstrate the existence of suitable design margins for environment and time domains. The particular characteristics to be tested are determined by the failure mode analysis, which is to be performed on every component used in the Apollo spacecraft. The more critical modes will be analyzed by this testing technique.

Control by Criticality

The logic employed to derive numerical values for criticality and the functional relationship between criticality and test time is predicated on the following assumptions:

1. Product integrity, as measured by the mean time between failure (MTBF), will increase as the amount of experience required from testing is increased; therefore, MTBF may be considered a direct function of the test time. This presupposes that testing reveals information that leads to design improvements or process or quality changes, all of which result in improved design integrity.

2. More test data are required on systems of higher criticality than on those of lesser criticality.

3. More test data are required on systems that cannot be improved through analytical techniques or those limited by state of the art.

4. Criticality is a function of the relative orders of redundancy, alternate modes of operation, spares incorporated in a given system design, and their effect on crew survival or mission success. Therefore, test time on systems composed mainly of series

components must be greater than that required for systems with parallel components or where spares and/or alternate modes of operation are provided.

Classification of Criticality

Upon calculation, the criticalities of the spacecraft components were found to fall into three groups, each separated by one order of magnitude. Additional justification for grouping criticalities was found in the fact that items with similar order-of-magnitude values not only contained similar degrees of redundancy or alternate modes, but also were responsible for similar ramifications in crew survival or mission success. The criticality was therefore grouped into the classes as shown in Table 1-1 with the definitions and reasons given.

Table 1-1. Criticality Classification

Class	Index Range	Possible Effect on Mission	Functional Relation
1	0.8 to 1.0	Impair crew survival (catastrophic)	No redundancy
2	0.01 to 0.08	Impair mission succes or prevent successful abort	Dual redundancy
3	Less than 0.008	Maintenance problem or nuisance	Triple redundancy of some form or nonessential equipment

Minimum Qualification Program

A minimum qualification program should be established as a base point and anything less than this considered as a compromise to the objectives of the mission, since it will yield insufficient data for meaningful qualification of the systems. An effective program will depend on multiple usage of the hardware. Further, no random or unknown failures can be tolerated. All failures must be designed out of the system. The tests should be planned and scheduled carefully to insure that a previous test conducted on a component will not prejudice the results of a following test. Refurbishment of components and systems should be planned when multiple tests are scheduled and detrimental effects expected during the earlier phases of testing. The minimum qualification program includes two component or lower levels of

assembly for sequential single-environmental, explosion-proof, and electromagnetic-interference tests — one component or lower level of assembly for off-limit tests (the unique requirement of this test plan) and one subsystem for mission simulation life tests. Replacements are necessary on highly critical, limited-life, or other items that require refurbishment prior to completion of this test phase. Table 1-2 summarizes the minimum hardware utilization.

Table 1-2. Minimum Hardware per System

Test Phase	Number of Black Boxes and Lower Assemblies	Number of Systems
Phase A - Design proof		
Part 1 - Environmental	2	0
Part 2 - Off-limit	1	0
Phase B - Mission simulation and life	0	1*
Total	3	1*

*Plus replacements on critical and limited life items; refurbish items requiring such action

Requirements for hardware, over and above this minimum program, were justified on the basis of criticality, expected mission life, and service life. The minimum program is applied to criticality 3 components only.

Test Time

The operating hours accumulated on a system from its fabrication to recovery were estimated by considering every step a component undergoes. A mission profile was drawn, depicting the prior-to-launch phases and the postlaunch phases. From the mission profile, individual systems or function "operated time" for prelaunch and "duty cycle" for postlaunch were derived. The total operating time to be imposed during each mission life cycle on a given system is the sum of both.

The time required to qualify a system under the mission-life concept was determined to be a function of its criticality to the system and/or the

total operating time accumulated throughout its expected life. The higher the criticality index, the more test data are required; and the higher the operating hours expected in a cycle, the more test time is required to qualify the system. Much of the operational time required to qualify under the mission-life phase will be derived from attribute data obtained from many other phases of the Apollo test program.

GSE Considerations

Minimum Qualification Program

There are many categories of tests, classes of GSE, environmental and criticality considerations, and functional groupings of equipment that must be organized into a comprehensive test plan with firm and logical criteria established to make each classification distinct and useful. Table 1-3 illustrates the factors and their interaction to be considered.

Table 1-3. GSE Classifications

Classifications	Functional Grouping	Categories
A. Equipment Class I, Mission essential Class II, Mission nonessential B. Environmental Class I, Sheltered Class II, Unsheltered C. Criticality Class I, Prejudice crew survival Class II, Cause pad or In-flight abort Class III, Cause count-down recycle	1. Checkout 2. Auxiliary 3. Servicing 4. Bench maintenance 5. Handling 6. SCD's	A. Development B. Qualification 1. Design proof 2. Life C. Acceptance

GSE, initially, was divided into mission essential and mission nonessential equipment, determined on the basis of its individual contribution to the mission. This division placed about 10 percent of the GSE in Class I

(MEE) and about 90 percent in Class II (MNEE). A criticality classification determined by analysis of effects on crew survival and mission assignments was then imposed on mission essential equipment to establish maximum and minimum levels of MEE qualification consistent with reliability requirements and the Apollo spacecraft test program. For the same reason, an environmental classification was imposed on the complete GSE to reduce the number of environmental test types required. Finally, a functional grouping of the GSE was made to complete the requirements analysis and assure the most effective test effort. Of the functional groups, the MEE classification is limited to the first three groups.

Although the terminology may be self-defining, mission essential will be defined here to promote understanding of the GSE qualification program. A part of GSE will be classified as mission essential if it is in a closed loop with the spacecraft anytime just prior to or during countdown, so that is may be involved in one of the following:

Fail to detect a malfunction

Introduce a malfunction

Cause a complete countdown recycle, short delays not included

Similarly, the classification of environment should be defined. The GSE is expected to be subjected to two classes of environments as a function of equipment use and location. Sand, dust, and salt fog are applied only to equipment expected to be unsheltered or in an uncontrolled environment. As in the spacecraft tests, it has been assumed, with NASA concurrence, that the equipment will receive special handling.

Test Items and Time

The sample size is limited in most cases to one item. In some cases, that item is also the end item to be delivered. Additional items will be required for certain mission essential equipment, although these are not included in present plans. The time available for qualification testing is limited to such an extent in some cases that only a conventional acceptance test can be accomplished.

The environmental limits applied to the GSE are very different from the spacecraft applications. The special handling decision resulted in a program basically limited to those specified in MIL-E-4970, and this applied only to those components that undergo a comprehensive qualification program. One-of-a-kind items are considered special cases and are evaluated by an expanded acceptance test only.

Implementation

Ground Qualification Tests. The ground qualification tests have been separated into two areas of responsibility. The first area and the first to be performed is the subcontractor's system qualification test program. The second area, following the subcontractor's effort, is the S&ID combined system and integrated system testing.

Subcontractor Qualification Test Program. This plan has been established in two phases and conducted at two levels of assembly. Phase A, Design Proof Tests, are conducted at the component level and Phase B, Mission Simulation Life Tests, are conducted at the system level.

S&ID Combined Systems and Integrated System Test Program. The ground qualification tests are completed by the progressive assembly of the separately qualified systems into an integrated Apollo spacecraft system for the verification of the dimensional and functional system interfaces. Interacting subsystems are tested first in combined systems tests to assure compatibility before commencing the integrated system test. House spacecraft will be used for these tests, including Airframe 008 intended for environmental proof testing. All data derived will be integrated by Reliability for system qualification and continual assessment of reliability.

Test plans are not firm at this reporting date. They will be reported as schedules materialize.

STATUS OF ACCEPTANCE TEST PLAN

The Acceptance Criteria, Volume IV of the General Test Plan Research and Development for Project Apollo spacecraft (SID 62-109-4), was prepared and released in March 1963. The intent of this document is to present a properly planned acceptance test program that will maintain the standard achieved by Engineering and preserve the inherent reliability of the qualified units from being degraded by the production process, material impurity, and poor workmanship. The scope of Volume IV includes the acceptance test program for the Apollo spacecraft system and its associated ground support equipment at the supplier's and S&ID's facilities. This is the first release of the document; therefore, it is not considered complete and will be revised during the next quarter. It does delineate the basis for the program and the philosophy of application (control at all levels parts, subassemblies, components, systems, and interfaces). Subsequent issues of this volume will present a more complete picture of the concept and application of the program.

The concept of the acceptance program is based on the premise that a rigorous control and the limitation of measureable quantities can attain the inherent reliability of the design. One of the prime requirements for this control is that the supplier acceptance test program be monitored by S&ID Quality Assurance personnel.

Also presented in this volume are the techniques in performing acceptance tests and the system acceptance test requirements. The techniques presented include functional, vibration, X-ray, infrared, ultrasonics, and magnaflux, and the requirements specified include all the known parameters to be checked during the acceptance tests.

QUALIFICATION TEST CRITERIA STATUS

The Apollo qualification test criteria were finalized during this quarter. It includes the qualification test criteria for both the space vehicle and ground support equipment. These data will form the basis for both the "Specification Input for Subcontractors and Suppliers Quality Assurance Provisions" and "General Test Requirement for Apollo Subcontractors and Suppliers" documents, which will be released during the next quarter. The criteria specified are divided into the major categories: design proof, off-limit, and mission life.

Also given detailed consideration were the test levels associated with the abort, reentry, and landing conditions. This is particularly significant in the dynamic (vibration, acoustics, acceleration, and shock) and the thermal environments. Due to the fact that some environments (thermal and acceleration-abort) are location dependent, these levels were not fixed. The levels that will apply in these instances will be determined at the time the equipment procurement specification is written and will be based on the specific situation relevant to the component in question.

The test criteria were developed and established to insure an adequate design margin relevant to each environment. Sufficient pad has been required to insure survival in the anticipated environment but not destroy the test article until off-limit tests are imposed. The off-limit steps will verify design margin.

II. SPACECRAFT ANALYSIS ACTIVITIES

SUMMARY

During this period, inputs have been completed for use in the reliability simulation model in preparation for spacecraft crew safety evaluation. Crew safety analyses are to establish reliability requirements based on the LOR mission, and these requirements will be verified for mission success. The analyses will also be used to define redundancy and on-board spares necessary to meet the requirements. The redundancy and spares will be chosen so that the greatest reliability gain per pound of added weight can be achieved.

A coordination meeting was held with representatives from Grumman Aircraft and MIT to establish and define areas where equipment within the LEM and the spacecraft could be used to support each other. Once these areas have been finalized, expected gains in reliability that would result from their implementation will be estimated. Design studies will then be conducted to determine the feasibility of implementing these designs.

ANALYSIS

Because of the complexity of the Apollo mission and spacecraft systems, a sophisticated reliability analysis technique is necessary. A trade-off study between the various applicable reliability disciplines was presented in SID-62-557-4. The Monte Carlo mission simulation model was chosen for the anayses of mission success and crew safety reliability.

Since crew safety is inherent in a successful mission and the crew safety objective can be achieved only through considerations of mission abort and safe return of the crew, the probability of crew safety (R_{cs}) becomes the sum of the mission success reliability (R_{ms}) and the product of the probability of abort (P_a) times the probability of safe abort (R_{sa}). The probability of abort (P_a) is defined as the probability of mission failure (Q_{ms}). Thus, the probability of crew safety

$$P_a = Q_{ms} = 1 - R_{ms}$$

can be expressed mathematically as

$$R_{cs} = R_{ms} + Q_{ms} R_{sa}$$

Because the existing Monte Carlo mission simulation model provided only the probability of mission success and mission failure, it was extended to include the analysis of the abort missions.

ABORT MISSION ANALYSIS

The probability of safe abort (R_{sa}) is a function of both the abort time in the planned mission and the abort mission profile to be followed in event of abort. The development of abort mission profiles describing the abort mission for those phases of the planned mission, therefore, was necessary where an abort was possible. The first step was to develop the nominal abort mission lengths, as shown in Table 2-1, to accompany the planned mission profile. Then, the abort mission profiles were defined as a function of the flight trajectory and the equipment operational requirements. A typical example is the profile of the abort mission from the transearch injection phase of the mission shown in Table 2-2.

The time of abort in the mission occurs when there is a mission failure. This time can be obtained from the mission analysis portion of the model.

Table 2-1. Apollo LOR Fourteen-Day Design Maximum Mission with Preliminary Abort Trajectories

Phase	Beginning of Phase Time	End of Phase Time	Phase Time	Minimum Return Time Impact (hr)
Launch to 100,000 feet	0	0.028	0.028	0.044166
100,000 feet to jettison of LET	0.028	0.042	0.014	0.234722
LET jettison to earth orbit injection	0.042	0.200	0.158	0.486722
Earth orbit stay (3 orbits)	0.200	4.700	4.500	0.486722
Translunar injection	4.700	4.780	0.080	0.486722
Translunar coast cisearth and LEM transfer	4.780	5.650	0.870	3.87
First midcourse correction	5.650	11.650	6.000	4.07
ΔV application	11.650	11.65125	0.00125	8.77
Translunar coast period 1	11.65125	20.65125	9.000	8.77
Translunar coast period 2	20.65125	29.65125	9.000	17.26825
Second midcourse correction	29.65125	35.65125	6.000	23.76985
ΔV application	35.65125	35.65250	0.00125	27.2705
Translunar coast period 3 and navigational sightings	35.65250	40.6875	5.035	27.2705
Translunar coast period 4 and navigational sightings	40.6875	45.7175	5.03	29.9696
Translunar coast period 5	45.7175	50.7475	5.03	32.4698
Translunar coast period 6	50.7475	55.7775	5.03	34.6685

Table 2-1. Apollo LOR Fourteen-Day Design Maximum Mission with Preliminary Abort Trajectories (Continued)

Phase	Beginning of Phase Time	End of Phase Time	Phase Time	Minimum Return Time to Earth Impact (hr)
Third midcourse correction	55.7775	61.7775	6.00	54.9702
ΔV application	61.7775	61.77875	0.00125	50.968
Translunar coast period 7 and navigational sightings	61.77875	68.27875	6.500	50.9668
Translunar coast period 8	68.27875	74.77875	6.500	46.47555
Fourth midcourse correction	74.77875	80.77875	6.000	41.97045
ΔV application	80.77875	80.780	0.00125	37.97125
Translunar coast period 9 and IMU alignment	80.780	81.780	1.000	37.97
Lunar orbit injection at 80 naut mi	81.780	81.870	0.090	37.47
Lunar orbit navigation period 1	81.870	83.700	1.900	71.51
Lunar orbit coast	83.700	122.700	39.000	71.51
Crew trans. and checkout	122.700	123.700	1.000	71.51
Injection into "equal period orbit" and transfer to 50,000 feet	123.700	124.200	0.500	71.51
Equal period orbit stay	124.200	129.850	5.650	80.18
Perilune retro (50,000 to 1,000 ft), hover, translation and touchdown	129.850	129.990	0.140	80.18

Table 2-1. Apollo LOR Fourteen-Day Design Maximum Mission with Preliminary Abort Trajectories (Continued)

Phase	Beginning of Phase Time	End of Phase Time	Phase Time	Minimum Return Time to Earth Impact (hr)
LEM lunar surface stay period 1	129.990	145.490	15.500	81.35
C/M navigational sighting	145.490	146.490	1.000	81.35
LEM lunar surface stay period 2	146.490	161.990	15.500	81.35
Lunar ascent to circular orbit at 50,000 ft	161.990	163.030	1.04	81.35
Rendezvous and docking	163.030	170.730	7.700	80.18
Lunar orbit coast period 1	170.730	207.955	37.225	71.51
Lunar orbit coast period 2	207.955	245.175	37.220	71.51
Lunar orbit navigation period 2	245.175	250.090	4.915	71.51
Transearth injection	250.090	250.130	0.040	69.514
Transearth midcourse correction 1	250.130	257.979	7.849	69.474
ΔV Application	257.979	257.98025	0.00125	62.475
Transearth coast period 1	257.98025	267.98025	10.000	62.47375
Transearth coast period 2	267.98025	277.98025	10.000	54.47375
Transearth midcourse correction 2	277.98025	283.98025	6.000	46.47375
ΔV Application	283.98025	283.9815	0.00125	41.87375
Transearth coast period 3	283.9815	288.9815	5.000	41.8725

Table 2-1. Apollo LOR Fourteen-Day Design Maximum Mission with Preliminary Abort Trajectories (Continued)

Phase	Beginning of Phase Time	End of Phase Time	Phase Time	Minimum Return Time to Earth Impact (hr)
Transearth coast period 4	288.9815	297.1335	8.152	37.7725
Transearth coast period 5	297.1335	305.2855	8.152	30.9725
Transearth coast period 6	305.2855	313.4375	8.152	24.6725
Transearth midcourse correction 3	313.4375	319.4375	6.000	18.4725
ΔV Application	319.4375	319.43875	0.00125	13.4725
Transearth coast period 7	319.43875	324.43875	5.000	13.47125
Transearch coast period 8	324.43875	329.43875	5.000	9.47125
Transearth midcourse correction 4	329.43875	335.43875	6.000	5.47
ΔV Application	335.43875	335.440	0.00125	0.97
S/M separation and reorientation	335.440	335.940	0.500	0.97
Entry	335.940	336.273	0.333	0.47
Deploy drogue chute through landing	336.273	336.410	0.137	0.137

The normal approach for analyzing the probability of safe abort is to continue simulation of the abort mission when a mission failure occurs. However, to achieve significant confidence in the Monte Carlo technique, many missions must be simulated, but since the probability of mission failure is relatively low, the number of mission simulations necessary to generate a statistically significant number of abort missions would require a prohibitive expenditure of IBM computer hours.

Table 2.2. Profile of Abort Mission from Trans-Earth Injection Phase

Mission Success Phase 39		
Abort Function	Start Time	Δt
ΔV firing	0	0.040
Deep space navigational sighting	0.040	2.283
Coast	2.323	4.566
IMU alignment	6.8890	0.500
Thrust vector alignment	7.3890	0.500
ΔV firing	7.8890	0.00125
IMU alignment	7.89025	0.500
Deep space navigational sighting	8.39025	2.500
Coast	10.89025	5.000
Deep space navigational sighting	15.89025	2.200
Coast	18.09025	4.400
Deep space navigational sighting	22.49025	1.667
Coast	24.15725	3.333
IMU alignment	27.49025	0.500
Thrust vector alignment	27.99025	0.500
ΔV firing	28.49025	0.00125
IMU alignment	28.4915	0.500
Deep space navigational sighting	28.9915	1.200
Coast	30.1915	2.400
Deep space navigational sighting	32.5915	2.267

Table 2.2. Profile of Abort Mission from Trans-Earth Injection Phase (Continued)

Mission Success Phase 39		
Abort Function	Start Time	Δt
Coast	34.8585	4.533
Deep space navigational sighting	39.3915	2.100
Coast	41.4915	4.200
Deep space navigational sighting	45.6915	1.733
Coast	47.4245	3.567
Deep space navigational sighting	50.9915	1.667
Coast	52.6585	3.333
IMU alignment	55.9915	0.500
Thrust vector alignment	56.4915	0.500
ΔV firing	56.9915	0.00125
IMU alignment	56.99275	0.500
Deep space navigational sighting	57.49275	1.000
Coast	58.49275	2.000
Deep space navigational sighting	60.49275	1.000
Coast	61.49275	3.050
Deep space navigational sighting	62.54275	1.667
Coast	64.20975	3.333
IMU alignment	67.54275	0.500
Thrust vector alignment	68.04275	0.500
ΔV firing	68.54275	0.00125

Table 2.2. Profile of Abort Mission from Trans-Earth Injection Phase (Continued)

Mission Success Phase 39		
Abort Function	Start Time	Δt
S/M separation	68.544	0.010
Orientation for Reentry	68.554	0.490
Entry	69.044	0.333
Deploy drogue chute through landing	69.377	0.137
	69.514	

The alternate approach used was to simulate many abort missions for each mission failure. The number of abort missions (K) simulated per mission failure is a function of the desired confidence interval in the resulting crew safety objective, and it can be expressed mathematically as follows:

$$A_{cs} = Q_{ms} A_{sa} + Q_{sa} A_{ms} = \sigma_{cs} \sqrt{\frac{R_{cs}\,(1 - R_{cs})}{M}}$$

where

A_{cs} = Accuracy interval of crew safety

A_{sa} = Accuracy interval of safe abort

A_{ms} = Accuracy interval of mission success

Q_{sa} = Probability of abort failure

σ_{cs} = Normal deviate of crew safety

M = Total missions simulated

The binomial distribution was approximated with the normal distribution because a large number of trials will be used. This approach defines the confidence that the true probability of crew safety (R_{cs}) is within plus or minus the accuracy interval (A_{cs}) of the observed probability of crew safety (R_{cs}). When these desirable parameters are known, the acceleration factor

(K) is then determined. Since the means of mechanizing the abort mission analysis is independent of the mission analysis, the mission analysis model was revised to provide the component failures in the mission at the time of mission failure. The failed components are then tabulated as failed components for the abort mission. Thus, the dependency of the abort mission on the planned mission is secured.

PROGRAM IMPLEMENTATION

The inputs to the crew safety portion of the Monte Carlo reliability analysis model are similar to those employed in the mission success portion presented in SID 62-557-4.

PROGRAM IMPROVEMENT

The program capabilities presented in SID 62-557-4 are now being extended through reprogramming in machine language and logic improvement. In addition, this effort will reduce the IBM computer hours required to process the program.

The maximum number of extended capabilities of the Monte Carlo mission simulation model are as follows:

1. 16,383 components per mission
2. 700 components to be used if simulation is necessary (This number includes all components related to components that may fail, i.e., alternate, combinatorial, dependent, and type 3.)
3. 500 functions simulated per mission and 700 functions in the crew safety master function list
4. 1,000 mission phases
5. 100,000 missions simulated per run
6. 5,000 function changes in the time function schedule
7. 200 functions that a component may support
8. 72 modes per function
9. 12 alternates that a component may have
10. 108 dependent components for any one component

11. 32,767 times that a component may be involved in a mission failure

This reprogramming effort will also include a revised input data program to simplify the input data. This will speed the reaction of the model to configuration changes.

GRUMMAN-NAA BACKUPS

The command module and service module subsystems that might receive either functional or equipment interchange backup from LEM are as follows:

Electrical Power
Cryogenic Storage
Environmental Control
Service Module Reaction Control
Service Propulsion
Crew Provisions
Guidance and Navigation
Stabilization and Control
Telecommunications
Instrumentation

This section is devoted to a discussion of these subsystems.

ELECTRICAL POWER

The interchangeability of Apollo and LEM electrical components and equipment could enhance over-all mission reliability. Present designs indicate that there is a similarity of components. To ensure equipment compatibility, where possible, criteria and requirements must be established during the early design and developmental phases of the program. The following information is presented as the present understanding of the Apollo electrical power system, and recommendations concerning significant interchangeability areas are given. Figure 2-1 shows the accessibility of these components for which interchangeability with similar LEM components is proposed.

Subsystem Description

The electrical power subsystem supplies, regulates, and distributes all electrical power required within the spacecraft during the mission. The major system components are as follows:

Three nonregenerative hydrogen oxygen fuel cells (primary power)
Three silver zinc batteries
One solid-state battery charger
Three solid-state a-c inverters
Two space radiator panels

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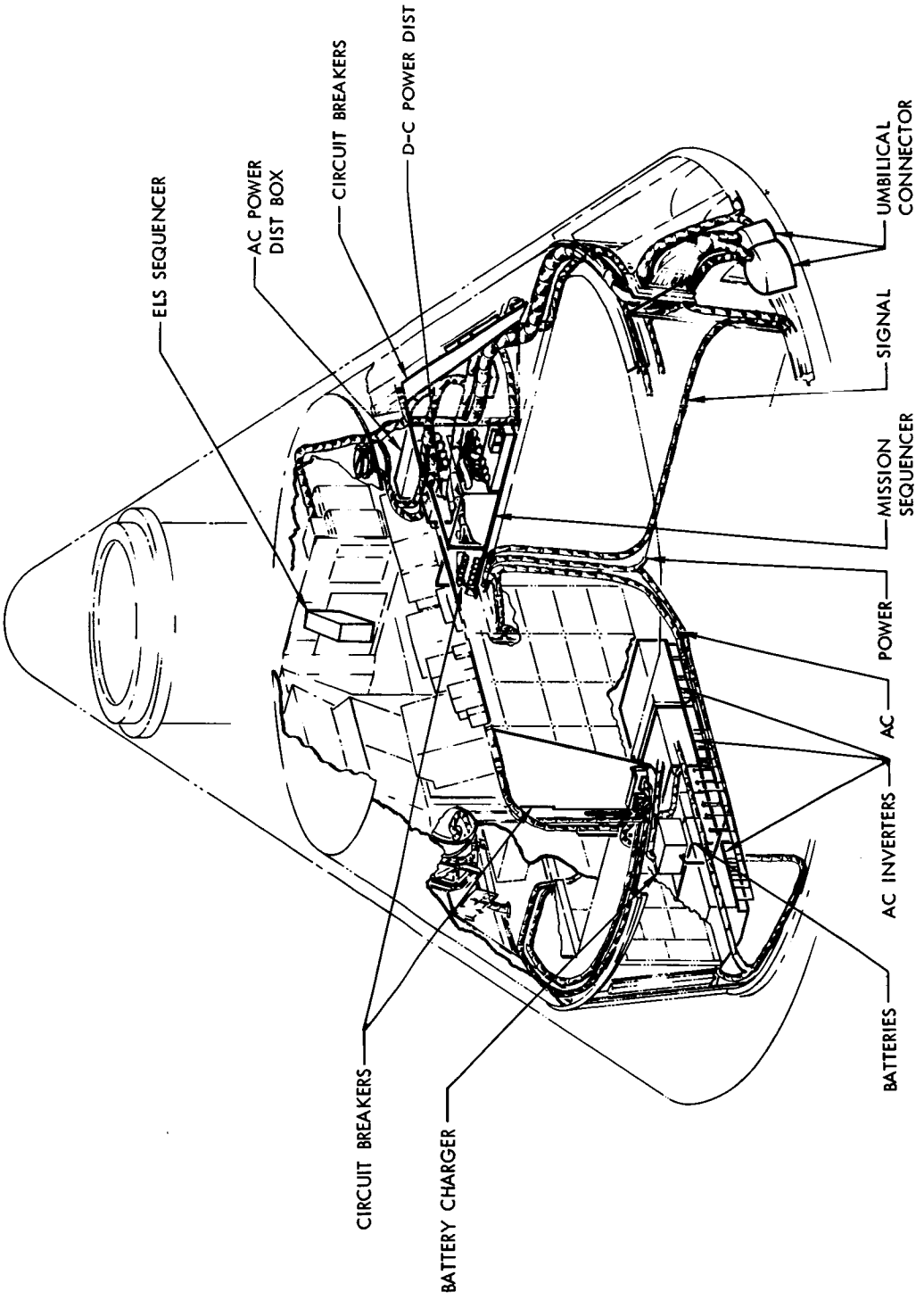

Figure 2-1. Command Module Electrical Components

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The component functions are the following:

1. Three fuel cells supply main power throughout the mission until reentry.

2. Two silver zinc batteries supply power during times of peak load, when operating in an emergency mode, and during entry and recovery.

3. One silver zinc battery supplies power during the postlanding phase. After earth landing, a 72-hour capability exists with use of all existing battery power and minimum postlanding load.

4. One battery charter is used to recharge all batteries prior to reentry, or as necessary.

5. One a-c inverter supplies 115-volt, 400-cycle power during all mission phases. Two inverters are maintained in standby.

6. A radiator panel dissipates waste heat generated in fuel cells.

The operation modes are:

1. Normal Mode. Three operative fuel cells supply power for all normal loads for a successful mission with one inoperative fuel cell, the remaining two modules supply power for all normal loads for a successful mission.

2. Emergency Mode. When two fuel cells are inoperative, all nonessential loads will be removed. The remaining single fuel cell supplemented by the reentry batteries supplies sufficient power to essential loads to permit return, reentry, and landing.

The system loads are as follows:

1. Essential. These loads are required to ensure the safe return of the crew under the emergency operating mode of the electrical power system.

2. Nonessential. These loads are not required to ensure the safe return of the crew under the emergency operating mode of the electrical power system.

The system parameters are the following:

1. D-C Power Characteristics

Characteristic	Rating
Steady-state voltage limits	25-30 volts (nominal 28 volts)
Normal transient voltage limits	25-30 volts for 0.7-second recovery
False condition transient (voltage limits)	21 volts for 1.0 second, 32 volts (0.7 seconds)
Ripple voltage	250 millivolts maximum, peak to peak
FC power	1500 watts, each (tentative)
Silver zinc battery life	25 ampere hours, 30 minutes, each (tentative)

2. A-C Power Characteristics

Characteristic	Rating
Three phase, displaced 120-2 degrees, WYE connected	
Steady-state voltage limits	115±2 volts root mean square, average three phases
Load 0-1250 volt-ampere, power factor varying from 0.7 lag to 1 unity	
Transient voltage limits	105-125 volts root mean square, 0.05 recovery
Voltage unbalance	2-percent maximum deviation of worst phase from average
Frequency limits	400+7 cycles per second, synchronized to spacecraft clock frequency

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Characteristic	Rating
Sine wave maximum total distortion; 5 percent, maximum individual harmonic and 4 percent, fundamental	

3. Power Limits

Mode	Limits
Normal	1.25 - 2.8 kilowatts
Emergency	0.9 - 1.35 kilowatts

Component Description

Fuel Cell

The single power plant consists of a fuel cell module with reactant control, water separation, pressurizing, and cooling provisions. The intermediate temperature, bacon-type fuel cells are for minimum reactant consumption at maximum power density and maximum heat rejection temperature. Each power plant is enclosed in a metal jacket. The water separation, heat transfer, and reactant control equipment is mounted in a compact package beneath the fuel cell.

Reactants are supplied to the module through a manifold system. They pass through reactant preheaters where some of the waste heat from the heat rejection section warms the reactants before they enter the pressure regulators. The pressure regulators maintain proper reactant pressure in the fuel cell, independent of the electrical demand. A 31-cell fuel cell, using dual-porosity nickel electrodes and operating at temperatures up to 500F, provides the d-c electrical power.

The reactants combine in the fuel cell to produce power, water, and waste heat. Potassium hydroxide (KOH) is used as a catalyst within the fuel cell. The water and waste heat are removed by recirculating hydrogen through the cell. The mixture of hydrogen and water vapor leaves the fuel cell and passes through the regenerator bypass valve. If the fuel cell temperature is too low, some of the hot gas passes through the regenerator to heat the cool hydrogen returning to the cell. Thus, the fuel cell operates at nearly constant temperature for all loads and environmental conditions.

Batteries

Entry Batteries. Two zinc, silver oxide batteries will supply all command module electrical power requirements during entry and recovery flight phases. In the event of one entry battery failure, the capacity of the other battery will be sufficient to supply essential loads during these phases. Entry batteries will also be used to supply peak system load demands during earlier flight phases. At all other times, the entry batteries will be isolated from load buses. One battery charge will recharge the entry batteries from fuel cell subsystem power.

Postlanding Battery. One zinc, silver oxide battery will supply essential loads during the postlanding phase. This battery will normally be fully charged and isolated from the main electrical power system. Provision will be made to recharge the postlanding battery in flight from the battery charger for emergency conditions such as excessive battery temperature. For normal postlanding conditions the unused capacity of the entry batteries will be made available for postlanding loads.

Battery Charger

One battery charger will charge the reentry batteries or postlanding battery, as required. During a normal mission, charging requirements will be minimal. Recharging, however, will be necessary prior to reentry.

Inverter

A-C inversion is done with a single, solid-state, three-phase, 400-cps 115/200-volt inverter. The operating inverter is supported by two idle standby units. The power rating of an individual inverter is 1250 volt-ampere.

Radiator

The radiator system is designed to be compatible with random radiator orientation in space for any seven-day period on the lunar surface and for all environmental conditions. The radiator for the fuel cell is located at the outer surface of the service module.

Electrical Power Distribution System

Load Classification and Power Source Operating Modes

All electrical loads supplied by the distribution system will be classified as essential, nonessential, pyrotechnic, or postlanding. Essential loads, except for pyrotechnic circuits, are defined as those mandatory for safe return of the spacecraft from any point in any mission. Nonessential loads

are those necessary for the successful completion of the mission. Pyrotechnic loads are essential loads, and postlanding loads are those required after earth landing.

D-C Power Distribution

The outputs of the three fuel cell modules and two entry batteries will be fed to the two isolated redundant essential buses, A and B. Redundant essential loads will be connected alternately to essential bus A or B. To retain bus isolation, nonredundant essential loads will be connected to both essential buses with isolation diodes. Redundant pyrotechnic buses will supply only pyrotechnic loads. All essential loads will be connected to the two essential buses so that loss of power to either bus will not cause interruption of power to any essential load system. The nonessential loads will be connected to a nonessential bus, and provision will be made for manually disconnecting these loads as a group under emergency conditions. The nonessential bus will be connected to both essential buses through isolating diodes. Postlanding loads will be supplied by the postlanding bus.

A-C Power Distribution

Three static inverters will provide a-c power by inversion from the spacecraft's primary d-c power sources. Transfer relays will manually switch inputs and loads to one of the other inverters in event of inverter failure. Inverters will supply a-c buses 1 and 2, to which alternate essential redundant a-c loads will be connected. Provisions are made to disconnect and isolate one of these buses from the inverter and the other bus in the event of bus failure.

Power Returns and System Grounding

Power distribution from the power sources will be by a two-wire system for d-c loads and single-phase a-c loads and a four-wire system for three-phase a-c loads. The power return current will be through the second d-c wire (or single-phase a-c load wire) or the fourth a-c wire for unbalanced three-phase a-c loads, to a negative d-c bus and a neutral a-c bus. These two buses will have the only ground connection to spacecraft structure in the spacecraft primary a-c and d-c power system. This ground connection will not be interrupted by control or switching devices. The negative and neutral return connections to primary power in each load system will be isolated from structural ground and from each other.

Recommendations

To ensure the maximum gain in reliability for both mission success and crew safety, specific interchangeability criteria have been recommended by Apollo Reliability Engineering. These recommendations are:

1. The Apollo single inverter power rating is 1250 volt-amperes, and the LEM inverter will be rated at approximately 1050 volt-amperes. Since physical size will not increase appreciably with an increased capacity of 200 volt-amperes, the LEM system should use the Apollo inverter.

2. The battery capacity for the LEM is undefined. It was recommended however, that, if possible, the Apollo main batteries be used.

3. No requirement now exists for a battery charger in the LEM vehicle. If it is determined later that a charger is required, the possibility of using the Apollo charger should be considered.

4. In regard to fuses, lamps and other similar electrical hardware, it has been determined that cannibalization of the LEM upon return from the lunar surface is both desirable and feasible. All failed equipment of the Apollo spacecraft would be replaced with usable LEM equipment where possible. At this time, all failed Apollo equipment would be jettisoned.

These recommendations represent only the major areas of interchangeability and are not complete. During all design phases of both vehicles, the possibility of equipment interchangeability should be thoroughly investigated.

CRYOGENIC STORAGE AND ENVIRONMENTAL CONTROL

Several areas of similarity may exist between portions of the command module-service module integrated system and corresponding subsystems in the LEM. Judicious study should indicate several areas where interchangeability can be accomplished either by physical cannibalization of parts from either vehicle or by transfer of the fluid stored within systems of one vehicle to systems of the other vehicle. Recommendations for reliability enhancement of the command module-service module with LEM by backup are limited to consideration of only those subsystems and components that have either similar configurations or functions in both vehicles.

Because of the inherent difficulties and dangers, it is not considered practical to exchange cryogenic storage system components. Backup considerations between the vehicles in this area should be limited to fluid transfer only.

The environmental control subsystem and the associated waste management subsystem present possible areas for consideration. Components such as suit circuit compressors, water glycol pumps, cabin fans, cabin pressure regulators, supply oxygen pressure regulators, suit circuit demand pressure regulators, water tanks, and coolant reservoirs appear to be the most logical areas where component interchange may be considered. In some mission phases the LEM waste management subsystem (prior to lunar operations) can probably back up the command module waste management subsystem. Vacuum cleaners, blowers, and the fecal and urine receptacles and storage containers could also be fabricated as interchangeable items.

Cryogenic Storage

A schematic diagram of the oxygen cryogenic storage subsystem is shown in Figure 2-2. This subsystem consists of two storage tanks with associated heaters and pressure switches that maintain the oxygen at critical pressure by providing added heat for compensation of fluid dissipated from the oxygen tanks. A distribution system is also provided for each storage tank. If either tank or its distribution system fails, the Apollo has enough fluid in each tank for safe abort to earth. Six hundred forty pounds of oxygen, stored at a nominal pressure of 900 psia is provided in the two tanks. Oxygen is also required for Apollo crew metabolic use and command module pressurization.

The hydrogen storage system is schematically quite similar to the oxygen system. Only 56 pounds of hydrogen, however, are stored at a nominal 250-psia pressure, compared with 640 pounds at 900-psia pressure for the oxygen system.

The hydrogen is only used as a fuel cell reactant. Backup between the LEM and the Apollo spacecraft does not appear feasible for the cryogenic storage subsystem. The great difficulties that would be encountered to transport either a storage container or the fluid stored at cryogenic supercritical pressures are not easily solved. If further study indicates that sufficient gains can be achieved by supporting the LEM with cryogenic fluid stored in the Apollo spacecraft or, conversely, by supporting the Apollo spacecraft with fluids stored in the LEM, perhaps a means of pumping the stored fluids from vehicle to vehicle should be developed. Only an intensive study, starting immediately, would indicate whether or not such a program should be undertaken.

The independent oxygen and hydrogen storage systems are considered to be in series for the first 147 hours of the mission (from liftoff to rendezvous with the LEM on its return from the lunar surface). The return mission will be conducted in the crew safety mode (redundant systems). The individual storage and distribution systems are considered to be

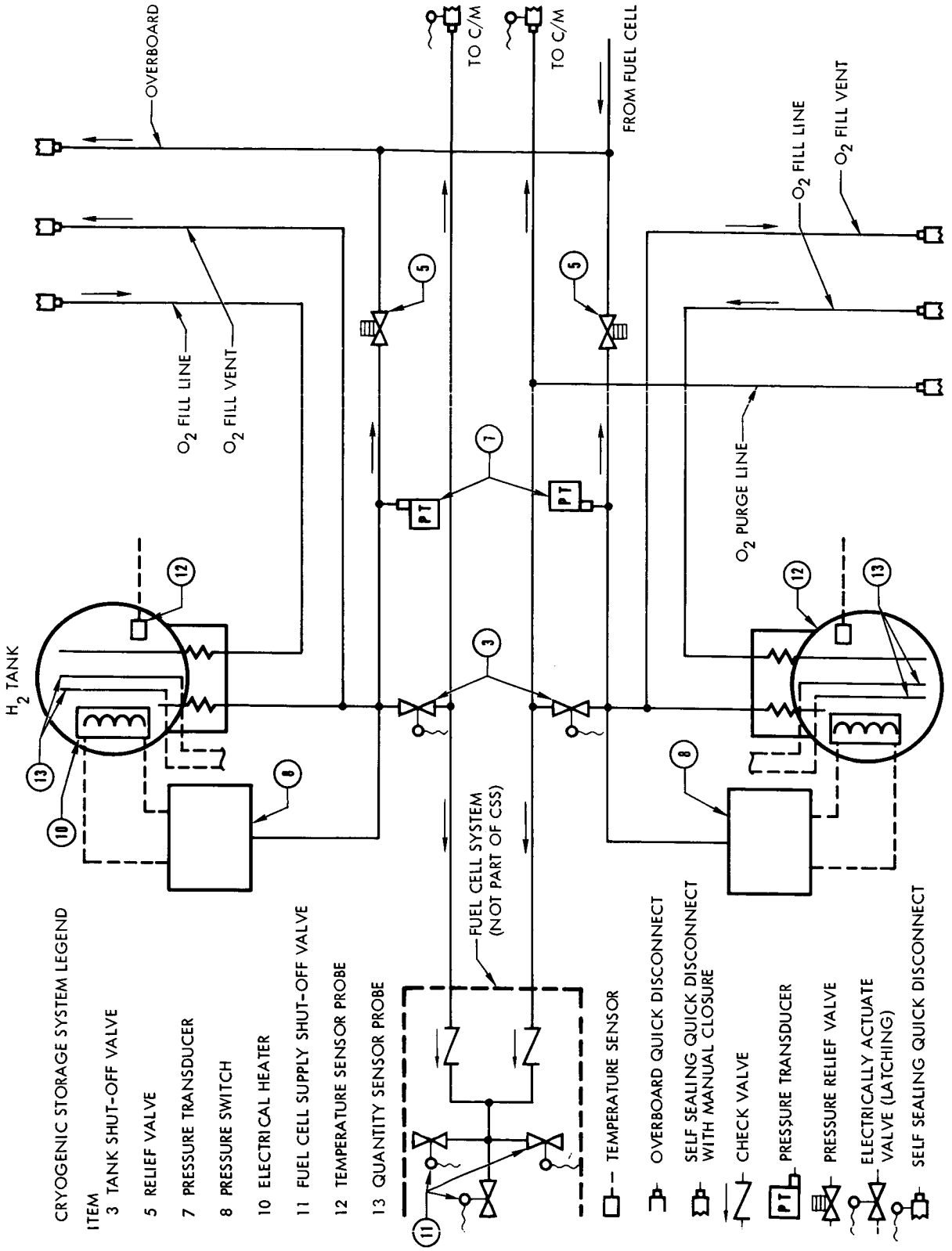

Figure 2-2. O_2 Cryogenic Storage

independent of each other. A single failure allows backup by the alternate storage and distribution system. Thus, any further backup, either in the crew safety or mission success modes of operation, could result in considerable reliability gains for the cryogenic storage subsystem. Of prime concern would be the possibility that, upon its return to the spacecraft, the LEM could provide its residual stored cryogenics to replace fluid lost from the Apollo spacecraft cryogenic subsystem. Thus, the mission could be extended, if desired, rather than terminated.

Environmental Control

The environmental control subsystem will provide conditioned temperature and pressure for crew shirt-sleeve habitation of the command module and oxygen for metabolic use. Figure 2-3 represents the accessibility of the major components.

The subsystem comprises five loops. A suit circuit provides oxygen for metabolic use that has been temperature conditioned, dehumidified, and treated for removal of odors and carbon dioxide. A water-glycol circuit acts as a heat transport medium for metabolic heat supplied by the crew and heat generated by on-board equipment. (Water glycol flows in a closed path from the command module to the service module where heat is rejected by space radiators, and the cool fluid is then returned to the command module where the cycle is repeated.) A pressure temperature control system provides a 5-psia atmosphere to the command module interior maintained at approximately 70 degrees F. An oxygen supply system supplies the command module suit circuit and pressure temperature control system with the oxygen necessary to satisfy metabolic requirements and replace oxygen lost by command module leakage. The water management system receives and stores the potable water generated by the fuel cells as a by-product and the metabolic waste water generated by the crew. The water stored in two tanks, potable and waste, is then available separately for consumption by the crew or for supplementary cooling during high heat loads in high thermal radiation environments.

The major redundant components are compressors, pumps, and fans. It is in this area where major gains in reliability and crew safety may be possible by mutual backup of the LEM and the command module, either by cannibalization of components or by transfer of operating fluids.

The major redundant components included in each system are self-explanatory, except for the oxygen supply system. The oxygen pressure regulator assembly indicated reduces the oxygen pressure supplied by the cryogenic storage system from the 900-psia stored pressure to 100-psia nominal environmental control system supply pressure. The demand pressure

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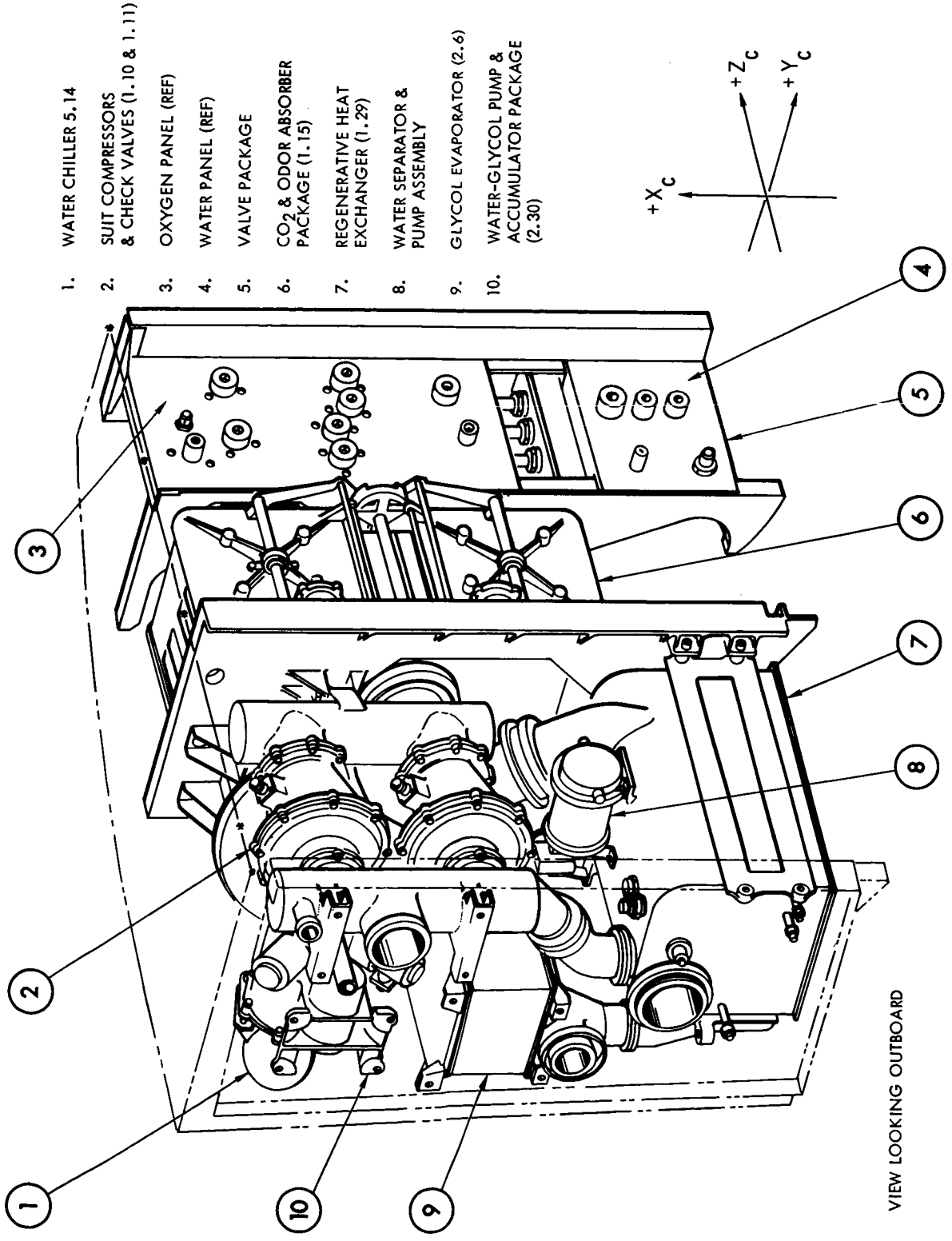
1. WATER CHILLER 5.14
2. SUIT COMPRESSORS & CHECK VALVES (1.10 & 1.11)
3. OXYGEN PANEL (REF)
4. WATER PANEL (REF)
5. VALVE PACKAGE
6. CO_2 & ODOR ABSORBER PACKAGE (1.15)
7. REGENERATIVE HEAT EXCHANGER (1.29)
8. WATER SEPARATOR & PUMP ASSEMBLY
9. GLYCOL EVAPORATOR (2.6)
10. WATER-GLYCOL PUMP & ACCUMULATOR PACKAGE (2.30)
+X_c
+Z_c
+Y_c
VIEW LOOKING OUTBOARD

Figure 2-3. Equipment Installation—Environmental Control System Left-Hand Equipment Bay

regulator and relief valve assembly supplies the suit circuit with regulated 5-psia pressure.

Dual rotating components, compressors, pumps, and cabin fans - are provided. From a reliability consideration, three components would be preferable. Because of weight and center of gravity factors, however, only two are currently provided. Similar functioning components will be provided in the LEM. If any of the rotating components fail, it appears feasible by proper design that components could be exchanged between vehicles. This would allow continuation of the mission or enhance crew safety.

In addition, similar considerations could be given to the pressure regulator assemblies that appear to have generic failure rates approaching those of the rotating components.

At present, the command module-service module will supply the LEM with its initial water supply. If the spacecraft water supply system fails, it is possible for the water remaining in the LEM to be shifted to the Apollo spacecraft. This transfer could be by water filled container or by water return through the original connector between the vehicles.

Because of the Apollo and LEM interface that requires the Apollo to pressurize the LEM with oxygen and to supply the LEM with its initial water, design of these interfaces must not significantly degrade crew safety or mission success.

In addition, the returning crew of the LEM should be able to sample the atmosphere within the Apollo prior to reentering the vehicle in case the astronaut remaining in the Apollo is incapacitated. Thus, the crew of the LEM would have some idea what functions are necessary for safe transfer back into the Apollo.

The waste management subsystem ties in directly to the environmental control subsystem, and a failure in this associated system could result in a premature mission abort. Since the LEM must also contain a similar subsystem, this could be another area of mutual backup by the LEM and the Apollo.

Figure 2-4 illustrates typical development and possible production configurations for several major components of the environmental control system.

SERVICE MODULE REACTION CONTROL

The service module reaction control subsystem operates during a lunar mission, except during reentry, and provides attitude control (± Pitch, ± Roll, ± Yaw) and translational control (± X, ± Y, ± Z) for the spacecraft.

by controlling propellant entry through solenoid valves and injectors into a radiation-cooled combustion chamber with a parallel flow exhaust nozzle.

Minimum control requirements are currently established as any two quadrants of four. The degree of quadrant redundancy is as follows: four quadrants successful through LEM transposition and docking, any three quadrants successful through LEM rendezvous and docking, and any two quadrants successful through command module-service module separation.

Utilization of LEM reaction control and service module reaction control as backups for each other could enhance the reliability of the mission and the probability of crew safety. During the translunar phase, LEM reaction control subsystem could provide attitude control during emergency operation for immediate or lunar free return if the service module-reaction control subsystem become partially or completely inactive. LEM reaction control subsystem could also provide small ΔV increments for trajectory correction. If LEM reaction control subsystem engine operation is not desirable, propellant transfer to service module reaction control subsystem tanks could compensate for propellant exhaustion. LEM reaction control subsystem design could be made so that emergency disassembly of engines could be accomplished and any failed service module reaction control subsystem engines could be replaced.

During lunar orbit before LEM excursion, several emergency operations could be performed including attitude control by LEM reaction control subsystem for navigational sightings or during repair of service module reaction control subsystem, small ΔV increments for trajectory improvement to conserve service module propellants, propellant transfer to replenish service module supply, and component interchange.

The LEM should provide all rendezvous and docking maneuvers, if possible. Since all functions are required during docking (attitude and translation control), loss of any one function can be replaced by service module reaction control subsystem engine operation. This allows degradation of both subsystems past their individual capabilities without loss of mission or crew. This method will require close cooperation between the two operators.

Subsequent to LEM excursion and before LEM command module separation, several backup modes of operation should be done. Attitude control of the spacecraft should be with the LEM-RCS until propellant exhaustion. This would ensure maximum utilization of equipment and would minimize service module RCS operation. Should time in lunar orbit be shorter than required to exhaust propellants, propellant transfer should be made to the service module RCS for optimum propellant management. If equipment fails in the service module RCS, it should be replaced with LEM-RCS components. These modes of operation give the astronauts more flexibility in mission accomplishment and increase reliability.

Details of propellant transfer must be studied, but equipment is already available to accomplish this except for interconnecting lines, which could be flexible and easily stored in coils. Each end could be a mating quick-disconnect filling to the fill and drain line of each tank. After caps have been removed and the line connected, fill and drain valves can be opened to provide a direct opening from one tank to the other. Pressure differential can be provided in the right direction in many ways, including, allowing tank pressure to decay slightly by closing gas supply to tanks, designing LEM-RCS tanks with slightly higher operating pressure, or providing a manually-operated valve which overpressurizes LEM-RCS tanks or by-passes regulator second stage. The pressure inside the transfer line will stabilize its configuration. When propellant transfer is completed (as observed by residual measurement), the line can be disconnected and thrown away and the caps replaced over the exposed ports.

Component interchange presents more of a problem because of the possibility of fluid leakage. The most promising component for this operation is the rocket engine, since it is on the outside of the spacecraft. Redesign of the mounting might be necessary to accommodate both engines. Each engine connection would have to provide fuel, oxidizer, and electrical connections as well as a secure thrust mount. Linear motion of the engine could separate two propellant quick-disconnects and an electrical connector. The engine could be released manually with a rotary latch of the bayonet type. Positive separation of propellant lines would prevent mixing in confined spaces, but the quick-disconnect seal be carefully designed. The gain is large, however, because one engine replacement weighing 3 to 4 pounds could activate a system weighing 200 pounds.

SERVICE PROPULSION

The service propulsion subsystem must perform 20 firings for a total of 660 seconds, starting from S-II separation, through the lunar mission until transearth mid-course corrections. The subsystem will provide the following functions:

Mid-course velocity corrections
Lunar orbit injection and ejection
Lunar orbit plane changes
Recovery of disabled LEM
Abort separation from booster

The service propulsion subsystem is comprised of helium supply, propellant feed, propellant utilization, and engine subsystems. The subsystem will provide thrust vector orientation and ΔV for mid-course corrections, lunar orbit injection and ejection, and lunar orbit plane changes.

[redacted]

The helium pressurization subsystem supplies regulated helium to the propellant subsystem during operation and isolates the fuel and oxidizer vapors during coast. The propellant feed subsystem then supplies pressurized fuel to the engine control valves and propellants to the rocket engine.

The propellant utilization subsystem allows the crew to adjust the fuel to oxidize mixture ratio so that the propellants can be maintained at the proper ratio and reduce residual propellants to a minimum.

The rocket engine is operated by actuating the pilot solenoid valves. These valves admit pressurized fuel into the main engine valve actuators that open the main engine ball valves allowing propellant flow into the thrust chamber. Satisfactory operation of the engine is achieved with either engine valve cluster. Electrically operated gimbal actuators position the thrust vector by positioning the rocket engine around its throat.

The LEM propulsion subsystem capability could be utilized for service propulsion subsystem backup in the following mission phases:

1. Translunar. After transposition and docking (for crew safety), the LEM could make mid-course corrections for immediate or lunar free return.

2. Lunar Orbit. Before LEM excursion (for crew safety), the LEM could make lunar orbit adjustments and transfer propellants from the LEM to the service module.

3. Rendezvous and Docking. For crew safety, the service module SPS could be used for LEM chase in the event the LEM comes back disabled.

4. Lunar Orbit. After LEM excursion (for mission success and crew safety), the LEM could backup lunar orbit adjustments and transfer remaining propellants from the LEM to the service module.

LIFE SYSTEMS CREW EQUIPMENT

Table 2-3 shows those items that are included in the command module-service module as crew provisions and for which cannibalization from the LEM could provide backup modes:

Table 2-3. Life Systems Crew Equipment

Crew accessories

- Crewman map and manual set
- Map and manual case assembly
- Crewman logbook assembly
- Crewman lapboard assembly
- Hatch egress accessories
- Crewman in-flight maintenance tool set

Crew support and restraint equipment

- Crewman couch pad assembly
- Crewman restraint harness assembly
- Crewman rest station restraint assembly
- Crewman navigation restraint system assembly

Lighting equipment

- Primary area lighting fixture assembly
- Secondary area lighting fixture assembly
- Crewman portable light
- Crew compartment windows filter assembly

Terminal survival provisions

- Crewman survival equipment provision assembly

Personal equipment

- Pressure garment assembly
- Portable life support system
- Constant wear garment
- Constant wear bump hot
- Umbilical hose
- Crewman in-flight maintenance tool belt
- Shoe strap
- Oxygen high pressure charging valve assembly
- Oxygen recharge hose assembly
- Cabin oxygen disconnect

Table 2-3. Life Systems Crew Equipment (Continued)

Sustenance and associated equipment

- Food
- Food package
- Food mouth piece
- Water delivery assembly
- Work and food preparation shelf assembly
- Food probe

Waste management equipment

- Fecal canister assembly
- Fecal-emesis bag (set)
- Urine receptacle assembly
- Receptacle liner set

Personal hygiene equipment

- Personal deansing pads package
- Ingestible dentifrice package
- Personal shaver
- Personal deodorant pads package
- Facial and toilet tissue package

Medical equipment

- Emergency medical supplies and equipment assembly
- Bio-medical instrumentation assembly (sensors)
- Pressing set
- Clinical, physiological monitoring instrument set
- Injectable drugs pack
- Topical agents pack
- Oral drugs (pills) pack
- Personel radiation dosimeters (NASA)

Personal communications equipment

- Personal communications
- Personal headsets (microphones and earphones)

GUIDANCE AND NAVIGATION

For this subsystem, it was concluded that functional backup could be used in certain areas, that some spares could be interchanged, but that no cannibalization at the component level could take place. Functional backup occurs during LEM rendezvous with the spacecraft. This could occur during an abort following LEM launch and prior to LEM landing, or after the lunar stay if a failure of the LEM occurred during either time period. Further functional backup is possible in the event of a failure of the command module G and N system in the translunar phase of the mission, as the LEM could possibly accompany the command module in the abort mission as far as the entry phase. The G and N equipment aboard the command module could guide the LEM. Spacecraft radar could locate the LEM, and telecommunication equipment could transmit directions to the LEM.. A recent discussion with MIT and Grumman revealed that many of the on-board spares will be interchangeable between the spacecraft and the LEM equipment. The basic mechanization of the computers will not be identical. Many of the individual modules will be identical. This interchangeability will also be feasible in other electronic parts, including coupling display units and optics electronics. Modules within the LEM that fail prior to launch could be interchanged for a successful LEM mission. The inverse of this interchange — the transfer of LEM components to the command module — could enhance the probability of crew safety prior to jettisoning the LEM, prior to an abort during the translunar phase, or prior to transearth inject following LEM rendezvous.

In areas other than electronics, cannibalization could not be feasible because of the difficulty or impossibility of interchanging equipment. For example, the optics are not removable from either craft. While it is possible to interchange the inertial equipment, there may not be sufficient tools or time available to effect this interchange.

STABILIZATION AND CONTROL

The potential backups of this subsystem are similar to those for the G and N system. Functional backup could be achieved during LEM docking. Again, it appears feasible to interchange spare items in the electronics. However, if the rate gyros fail, the possibility of interchange could present problems because of installation difficulties.

COMMUNICATION AND DATA

Functional backup within this subsystem consists of relaying messages from the LEM through the command module equipment during the lunar operational phases. Cannibalization of equipment at the component level

appears to be feasible in this system. A certain amount of electronic spares interchange also appears to be feasible for these equipments. This will be predominantly in the audio sections, and VHF and S-band communication equipment.

INSTRUMENTATION

Equipment included in this subsystem are the antenna and the in-flight test systems. The only functional backups would be those covered in the Communication and Data subsystem. It may be feasible to interchange certain electronics within the in-flight test system. However, this cannot be evaluated until a more definitive design is developed for both spacecraft and LEM equipment.

ENVIRONMENTAL CONTROL

SUMMARY

Predictions for attainment of mission success and crew safety reliability goals with the present ECS configuration are summarized in Table 3-1. Mission success goals cannot be attained with the present design, but added redundancy for the suit compressors and water glycol pumps will allow these goals to be met. As failure rate improvements occur, crew safety goals are expected to be attained.

Problems presented in the first design review and reported in Figure 23 of the Fourth Quarterly Reliability Status Report, SID 62-557-4, were resolved through design review activity. Design studies were made of the feasibility of installing an additional water glycol pump and of carrying a spare suit compressor. Vendor management activity included visits to the supplier's facility to evaluate activity and to direct progress.

Table 3-1. Environmental Control System Apportionments and Predictions

Item	Mission Success Reliability		Crew Safety Reliability	
	Apportioned	Predicted*	Apportioned	Predicted*
Suit circuit	0.99949	0.9910	0.999931	0.99985
Water glycol circuit	0.999133	0.9919	0.999885	0.99987
Pressure and temperature circuit	0.999812	0.99932	0.999972	0.99997
Oxygen supply	0.999884	0.99957	0.999981	0.99998
Water supply	0.99949	0.99834	0.999931	0.99993
Environmental control system	0.997675	0.9805	0.9997	0.9996

* Based on genuine failure rate data for state-of-the-art parts

ANALYSIS

Failure Effects Analyses and Logic Diagrams

No major system changes occurred during the reporting period. Therefore, failure effects analyses and system logic diagrams presented in SID 62-557-4 (Figures 17 through 22 and Table 23) have not changed. Figure 3-1 presents a logic diagram that shows the redundancy available for the critical equipments.

Abort Criteria

The following abort criterion has been established to calculate mission success for systems essential to crew safety: mission abort will be effected when one more failure would result in crew loss. Abort, then, must occur following a single failure of an ECS pump or compressor. This requirement exists for only the first 147 hours of the 336-hour mission since the return to earth cannot be considered an abort.

Crew safety reliability is calculated by adding the probability of success up to a specific mission point in time (point from which the crew could turn around) to the probability of abort necessity multiplied by the probability of safe abort (i.e., returning to earth). This process is repeated for each phase from which a mission could be aborted, and the product of these calculations yields the total probability of crew safety. The generalized equation is as follows:

$$Pcs = (Pms_I + Qms_I \cdot Psa_I) \cdot (Pms_2 + Qms_2 \cdot Psa_2) \cdots$$

$$(Pms_N + Qms_N)\, Pms_{N+1}$$

where

Pcs = Total system crew safety reliability for the 14-day mission

Pms = Probability of mission success during a specific mission phase (e.g., earth orbit)

Qms = Probability of the occurrence of a failure requiring abort during that specific mission phase

Psa = Probability of successful return to earth following abort

Pms_{N+1} = Probability of successful completion of the last phases of the mission

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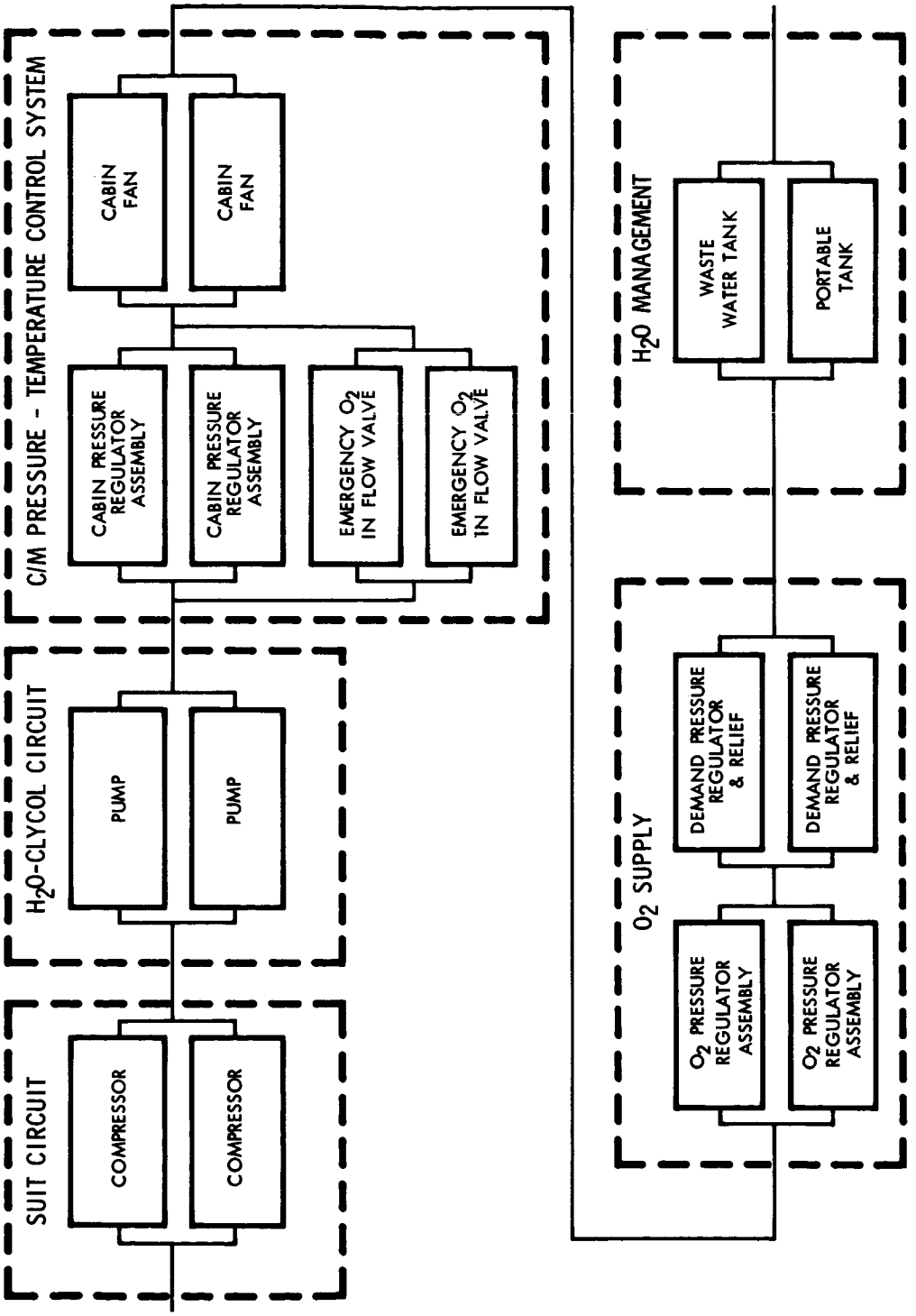

Figure 3-1. Environmental Control System Reliability Logic Diagram

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A rigorous derivation of the preceding equation is presented in SID 62-557-4 (pages 47 through 63).

Reliability/Crew Safety Design Reviews

No formal design reviews were held during the reporting period. Activity, however, continued on the action items reported in SID 62-557-4 (Figure 23). The following design review items received attention.

1. Rotating components with first-order failure modes, suit compressors, and water glycol pumps were investigated by the design group. Addition of a third water glycol pump to the present ECS package is under consideration. The possibility of carrying a spare suit compressor is being investigated.

2. Radiation effects on ECS materials were investigated, and the materials were found satisfactory.

3. The Design Review Board recommended a gas analyzer for monitoring the command module atmosphere. A NASA-supplied gas analyzer is to be used; in addition, CO_2 level will be sensed by subcontractor-supplied equipment.

4. Fire hazards of dye penetrant inspection fluids were evaluated, and these fluids were found to be satisfactory in 100-percent, 5-psia oxygen atmosphere, but they do constitute a hazard in high pressure and liquid oxygen environments. Therefore, for purposes of crew safety during possible malfunction, dye penetrants will not be used in the ECS. Meteoroid damage, up to a half-inch diameter hole, was investigated, and it was found that the CM oxygen system will supply O_2 at 3.5 psia for 5 minutes. This time is adequate for the crew to don space suits.

Solutions to the design review action item problems will be documented.

SPECIAL STUDIES

The Redundant Component Trade-Off Study of rotating components (suit compressor, glycol pumps, and cabin fans) completed during the previous reporting period (SID 62-557-4, pages 132 through 136) was reevaluated. As a result of this new study, the cabin fans have been removed from the first-order failure mode classification. A shirtsleeve atmosphere can be maintained after one fan failure. After the second fan failure, crew safety will be maintained with the pressure suit. Mission success goals can be attained only with a minimum of three suit compressors and three water-glycol pumps.

TEST PROGRAM

Qualification Test

Data was submitted for inclusion in Volume III (Ground Qualification Tests) of SID 62-109. The data included a definition of the system test program, a system logic diagram, a presentation of system criticality versus hardware requirements, and a qualification test schedule with test specimen utilization. Component qualification testing is scheduled to begin in June 1963.

Specific Development Test

Approximately 60 percent of the environmental control system is currently in the developmental testing stage. Leakage tests at high and low temperatures were satisfactorily performed on ECS control valves. Cycle tests were performed on the suit evaporator control valve (10, 000 cycles) and the glycol temperature control valve (2400 cycles) with no problems or failures. The regenerative heat exchanger bypass valve activator was successfully tested at ambient room temperature and under tension and compression for durations as follows:

Test	Hours
Room temperature	8
Elevated temperature	2
Hard vacuum test under pulsating operation at 10^{-4} Torr pressure	108
Simulated life test	100
Vibration test	9
Temperature soak test:	
High	48
Low	72

Reentry O_2 supply tank burst tests were performed satisfactorily. Tests utilizing a wick-type water separator operating in conjunction with a self-cycling water accumulator were run. The results proved the feasibility of design.

A prototype glycol pump with a full complement of bearings completed the 1000-hour (1073 actual hours) life test without failure or deterioration of performance. Pump performance efficiency and power consumption were better than expected.

The glycol system test is currently in progress. No major problems have been encountered; however, minor design improvements have been made on one or two valves.

Stability and control point accuracy of the cabin temperature control system has checked out acceptably. A 3 F union in the temperature control set point was corrected by replacing one of the control calibration resistors.

A prototype evaporator test rig was set up to check out the performance of the boiler with the associated temperature control and fluid valving. There was a need for minor redesign of the temperature control. The results showed flooding on water flow initiation and icing upon water flow cutback. Back pressure control is being used to see if it will prevent icing.

SUBCONTRACTOR MANAGEMENT

Subcontractor Coordination

Reliability Program Plan (Revision 2 of Reliability Program Plan)

SS-1003-R was prepared but never transmitted to NAA, pending resolution of Program Test Plan, SS-1001-R, Revision 2. However, following receipt of NAA Statement of Work, SID 62-10, dated 12 October 1962, and Change 1, dated 5 December 1962, new direction for reliability planning was available. As a result, Revision 3 of Reliability Program, SS-1003-R, dated 21 December 1962, was prepared and submitted to NAA on 28 December 1962.

This plan was conditionally approved on 4 March 1963 by Apollo Reliability for implementation by AiResearch. The approval was qualified by a requirement that AiResearch review various applicable reliability specifications and submit plans, documents, and procedures implementing or conforming to these specifications.

Coordination Meetings

A coordination meeting was held on 13 March 1963 at NAA-S&ID with representatives from AiResearch. Agenda items discussed were the following:

1. Status of the ECS reliability analysis pertaining to the E change
2. Predicted system mission success and crew safety reliability versus apportioned values
3. AiResearch recommendations for reliability improvements
4. Status of the reliability design review
5. Effects of instrumentation on over-all system reliability
6. Scope of system component compatibility testing
7. Testing to ensure crew safety after impact
8. Current test schedule
9. Mission essential GSE

During the meeting, NAA-S&ID stressed the importance of AiResearch reliability personnel participating in design changes proposed by their engineering department and exerting more influence in the design through reliability design analysis techniques.

Minutes of the meeting were published in Memorandum AREM 63-70, dated 20 March 1963.

Visits to Subcontractor Facility

Apollo reliability personnel have made visits to AiResearch's facility to evaluate reliability activities and to offer guidance and direction. During the visits, Apollo Reliability (NAA) stressed the importance of the AiResearch Reliability Group's influence on the ECS design, especially the crew safety items.

A design review meeting at AiResearch was attended by NAA-S&ID Apollo Reliability during the reporting period. The design review considered the glycol manual shut-off valve package 2. 27.

Subcontractor Activities

The following activities were reported to NAA by the AiResearch Reliability Department.

1. Preliminary logic diagrams were drafted and submitted informally to the NAA Reliability Group. Formal submittal of these diagrams will be made during the next reporting period. A prediction of ECS reliability has been accomplished in conjunction with the logic network effort. The effect of NAA mission time change from 400 to 500 hours will be evaluated.

2. Testing to date has been accomplished largely by the design groups to determine if design concepts are suitable for the application. These early tests have yielded little usable reliability information. ECS hardware is presently in the development test phase, and component qualification testing is scheduled to commence during June 1963. A qualification status list will be prepared approximately 30 days prior to testing.

3. Design reviews on early prototype designs have been completed. Design reviews are being conducted on the components comprising the 51175E ECS. Approximately 75 percent of the 51175E ECS components have been subjected to at least one design review. Work was initiated on failure mode and effect analyses applicable to the 51175E ECS. These analyses were scheduled to be completed during April 1963.

4. Trouble reporting has been maintained through the research and early development stages of the Apollo ECS design growth. These early tests were mainly exploratory to determine the feasibility of design concepts, and they yielded little reliability information. The ECS component designs are now approaching a development stage where production configuration will be reflected. When testing begins on this type of configuration, troubles will be reported verbally to NAA-S&ID resident representatives, and then written trouble reports will be transmitted to NAA-S&ID.

5. ECS component testing is passing from the preliminary development stage to the start of formal qualification testing. Detailed problems involving reliability, or critical design parameters, however, cannot be summarized at this time. For example, a minor bearing change increased the glycol pump life from 180 to more than 1000 hours.

PLANNED ACTIVITIES

ECS System Changes

Several design changes are being investigated to improve expected ECS performance and reliability. Included in the changes under investigation are installation of automatic H_2O-glycol flow control through the radiators, relocation of radiator shutoff valves for manual control in the CM, and revision of the water management system (Figure 3-2), which features a separate freon tank for supplementary cooling during launch and improves water accessibility for crew survival after reentry. These changes will be thoroughly analyzed for total effects on system reliability and crew safety. If they are adopted, updated logic diagrams, math models, and failure analyses will be prepared.

Associated Systems

The development of the water management system, radiators, and cold-plate circuits is approaching final configuration. Formal analysis will be completed to determine their effects on over-all ECS reliability. Preliminary results indicate that the reliability and crew safety of these systems, based on predicted failure rates, will match apportioned values.

Subcontractor Direction

As discussed under subcontractor management, the subcontractor's reliability program plan has been approved. The subcontractor will be under close surveillance for compliance with his plan. Strong direction will be given to the contractor in areas where his reliability efforts may not comply with this plan or are entirely lacking.

LEM Integration

ECS integration for function with the LEM and backup is expected to receive considerable attention.

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ITEM NO.	NO. REQD.	DESCRIPTION
1.27	1	SUIT EVAPORATOR TEMPERATURE SENSOR
1.28	1	SUIT EVAPORATOR TEMPERATURE CONTROL
1.29	1	HEAT EXCHANGER PACKAGE
2.22	1	GLYCOL TEMPERATURE CONTROL
2.23	1	GLYCOL TEMPERATURE SENSOR
2.6	1	GLYCOL EVAPORATOR
2.31	1	GLYCOL TEMPERATURE CONTROL VALVE
3.1	1	CABIN OUTFLOW, PRESSURE REGULATOR AND NEGATIVE RELIEF VALVE
5.1	1	WATER SUPPLY QUICK DISCONNECT & PRESSURE CAP
5.2	5	WATER CHECK VALVE
5.5	1	SUIT EVAPORATOR WATER INFLOW CONTROL VALVE
5.10	1	POTABLE WATER TANK
5.14	1	WATER CHILLER
5.15	1	WASTE WATER TANK
5.18	1	WATER SHUT-OFF VALVE
5.19	1	WATER TANK PRESSURE RELIEF VALVE
5.22	1	POTABLE WATER SUPPLY ASSEMBLY SUPPLY VALVE (2 WITH COMMON HANDLE) RESERVOIR WITH HEATER
5.23	2	WATER PRESSURE RELIEF VALVE
5.24	2	TANK PRESSURE CONTROL & RELIEF VALVE
	1	FREON STORAGE SYSTEM TANK
5.12		FILL DISCONNECT & PRESSURE CAP
5.25		FLOW CONTROL VALVE
	2	SELF SEALING QUICK DISCONNECT WITH PRESSURE CAP
	1	OXYGEN SHUTOFF VALVE

MANUAL OVERRIDE CAPABILITIES

5.19	ACTUATE TO CLOSED POSITION ONLY
5.23	ACTUATE INDIVIDUALLY TO CLOSED & OPENED POSITIONS
5.25	ACTUATE OPEN OR CLOSED
5.5	ACTUATE OPEN OR CLOSED BY INCREMENTS

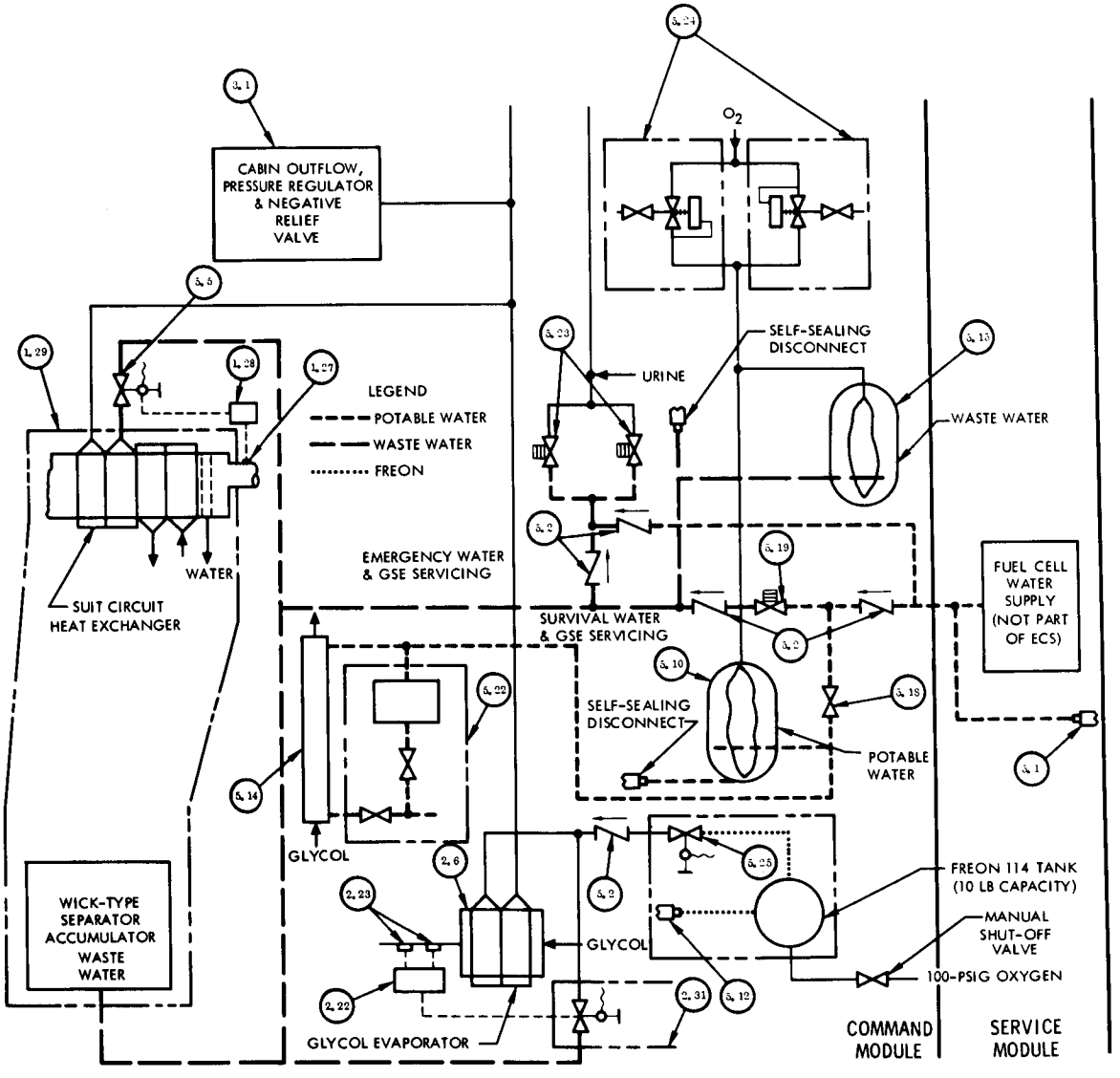


Figure 3-2. Revised Water Management System Schematic

ELECTRICAL POWER

SUMMARY

During the period covered by this report, major reliability effort on the electrical power system (EPS) was directed toward refining the logic network. The updated logic diagram shown in Figure 3-3 is based on the detailed failure effects analyses discussed on pages 102 through 130 of SID 62-557-4, Fourth Quarterly Reliability Status Report. Utilizing this diagram, a mathematical model for reliability of the EPS was developed (Figure 3-3). This is primarily a series-system model, but it involves all conditions under which the system will function. The model provides the reliability level and a means of identifying potential and actual reliability problem areas.

Recent use of the mathematical model has provided refined system and subsystem reliability values. Table 3-2 contains a summary of apportioned and presently predicted reliability values. Also indicated in this table are apportioned and present percentage of unreliability based on failure-rate data. Utilizing this information, it was possible to determine subsystems requiring the highest degree of design improvement effort. Data shown in Table 3-3 were used to derive these valves.

To provide a means of continual reliability surveillance, initial steps have been taken to develop reliability growth curves for all major components of the EPS. The analysis and resulting growth curves will facilitate the following:

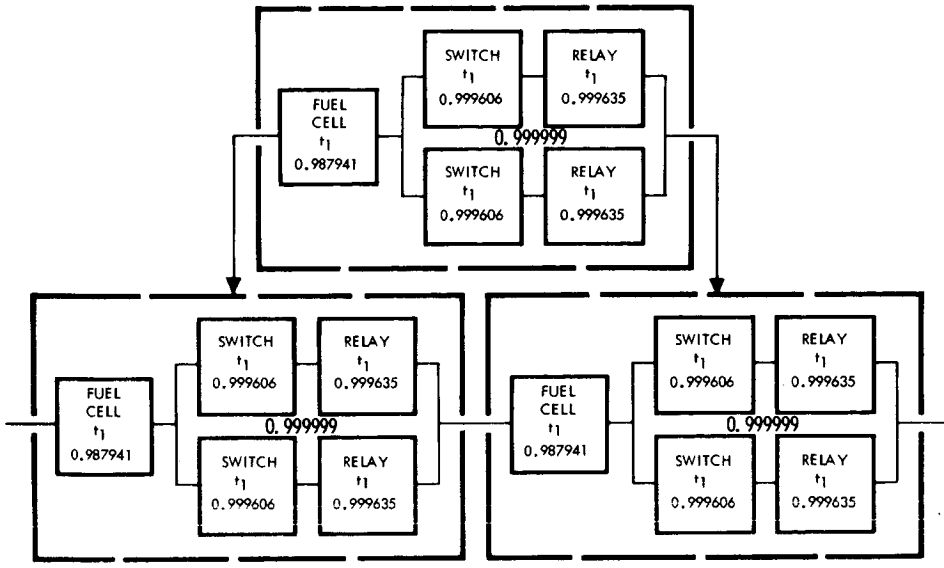
1. Prediction of system capability

2. Identification of potential problem areas

3. Indication of areas where improvement will be most effective

Predictions and assessments will be kept current throughout the program. As more information about the design and environmental effects on system components is made available, the model will be utilized for more accurate predictions and for analysis of design changes.

As designs are developed and test data acquired, failure-rate prediction will be supplemented by test data which will provide more accurate predictions of the EPS components and, consequently, the system itself. All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RELAY t_1 0.999635
0.999999
SWITCH t_1 0.999606
RELAY t_1 0.999635
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SWITCH t_1 0.999606
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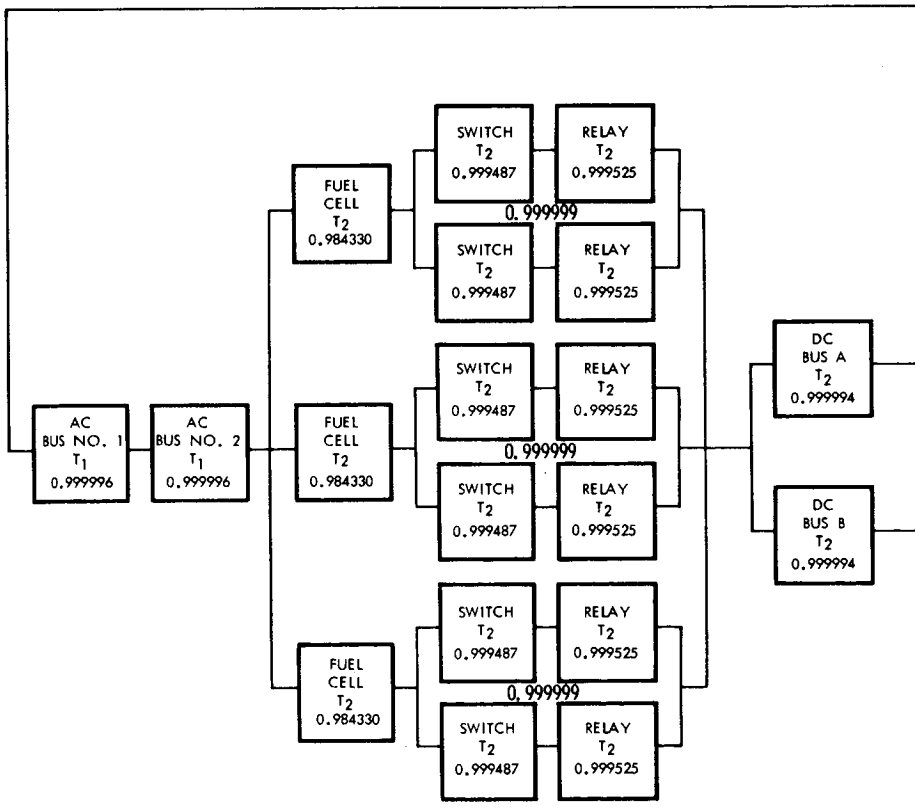
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AC BUS NO. 2 T_1 0.999996
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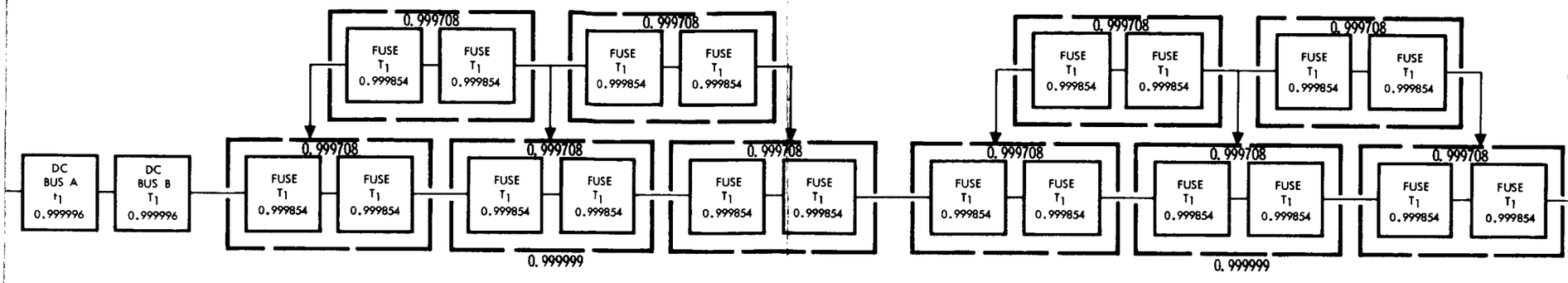
SWITCH T_2 0.999487
RELAY T_2 0.999525
0.999999
SWITCH T_2 0.999487
RELAY T_2 0.999525
FUEL CELL T_2 0.984330
SWITCH T_2 0.999487
RELAY T_2 0.999525
0.999999
SWITCH T_2 0.999487
RELAY T_2 0.999525
FUEL CELL T_2 0.984330
SWITCH T_2 0.999487
RELAY T_2 0.999525
0.999999
SWITCH T_2 0.999487
RELAY T_2 0.999525
DC BUS A T_2 0.999994
DC BUS B T_2 0.999994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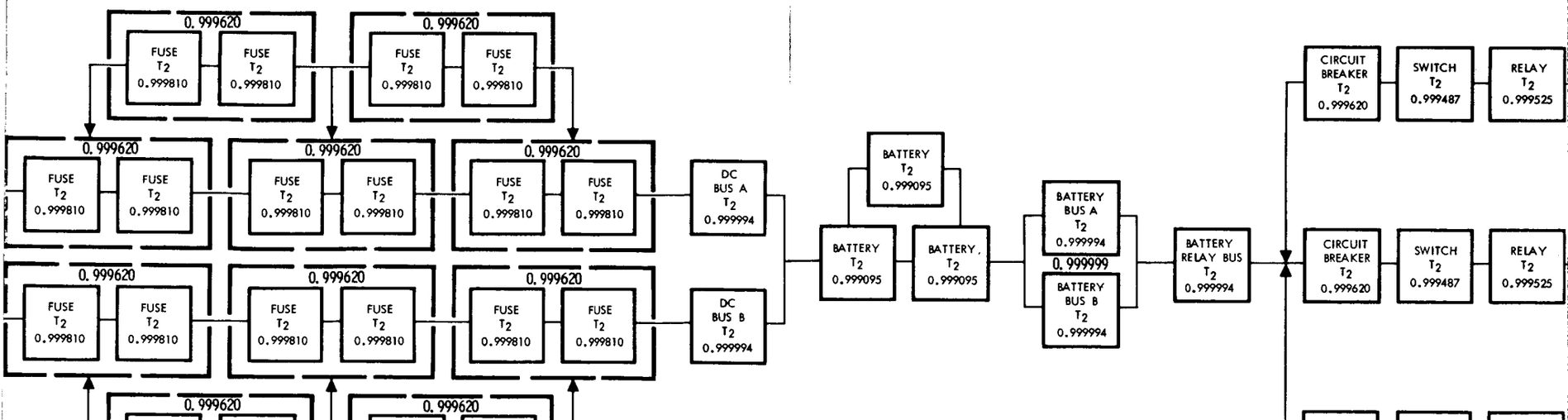
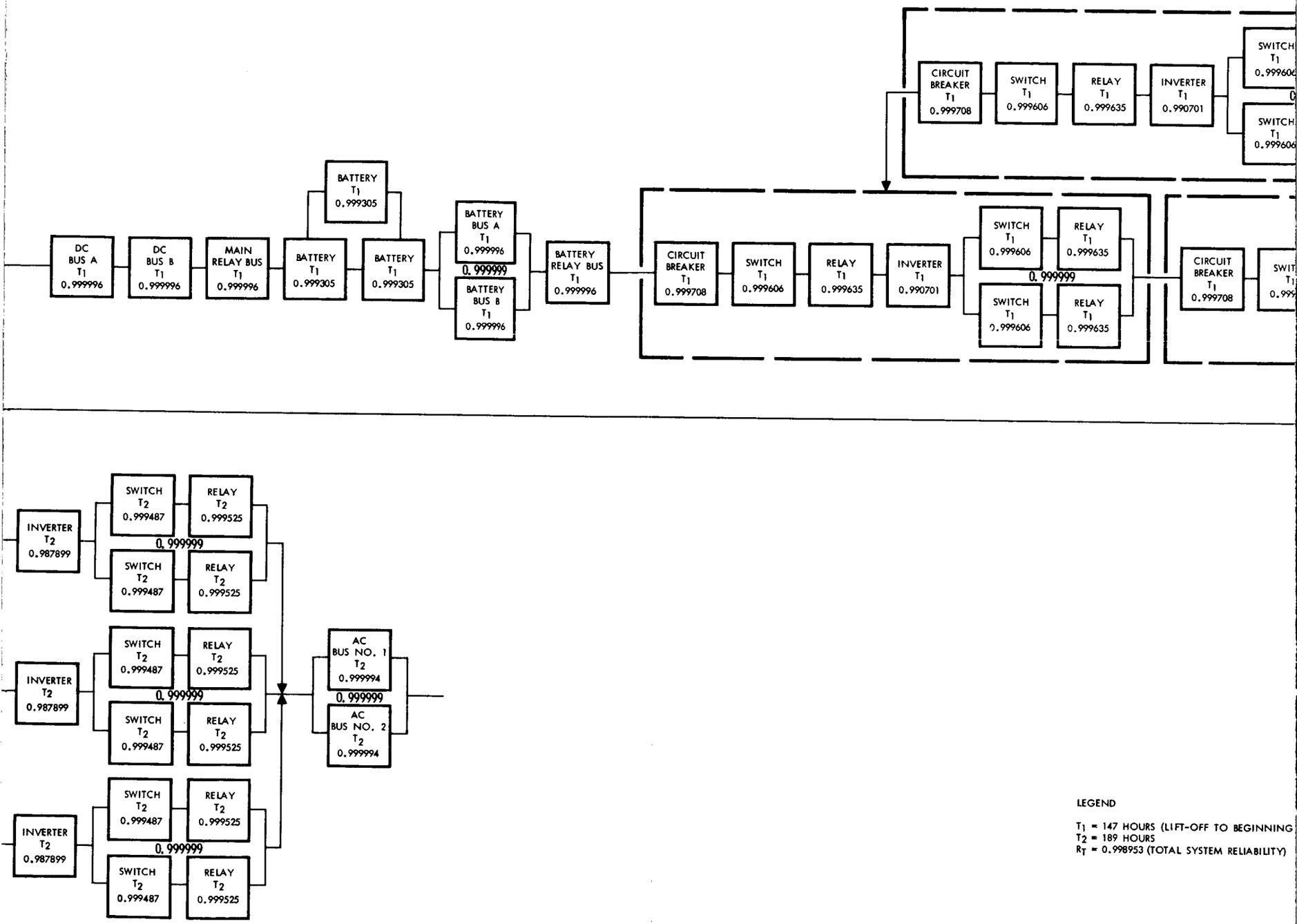
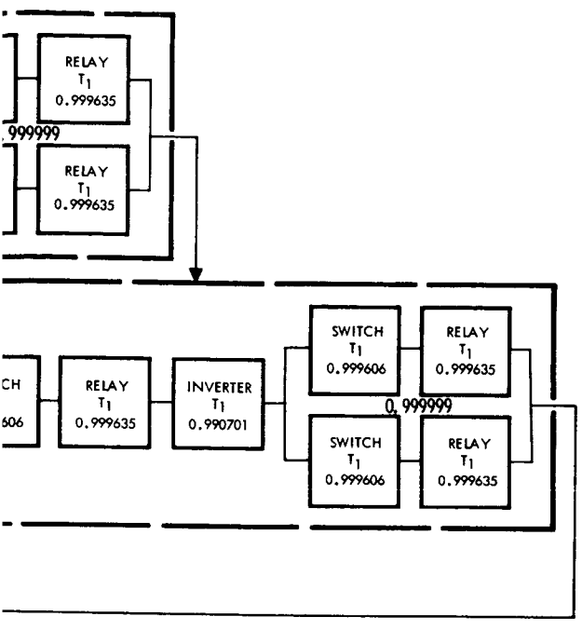
DC BUS B T1 0.999996
FUSE T1 0.999854
0.999708
0.999999


FUSE T2 0.999810
0.999620
DC BUS A T2 0.999994
DC BUS B T2 0.999994
BATTERY T2 0.999095
BATTERY BUS A T2 0.999994
BATTERY BUS B T2 0.999994
0.999999
BATTERY RELAY BUS T2 0.999994
CIRCUIT BREAKER T2 0.999620
SWITCH T2 0.999487
RELAY T2 0.999525



LEGEND

T_1 = 147 HOURS (LIFT-OFF TO BEGINNING
T_2 = 189 HOURS
R_T = 0.998953 (TOTAL SYSTEM RELIABILITY)



OF LUNAR SECOND-STAY PHASE)

ELECTRICAL POWER SYSTEM

t_1 = 107 HOURS — COMPLETE MATH MODEL — t_2 = 189 HOURS

$$R_{\text{MISSION SUCCESS}} = \left\{\left[R_{F/C}\left(1-\left[1-R_{SW}R_R\right]^2\right)\right]_{t_1}^3\left[\left\{R_{F/C}\left(1-\left[1-R_{SW}R_R\right]^2\right)\right\}^3+3\left\{R_{F/C}\left(1-\left[1-R_{SW}R_R\right]^2\right)\right\}^2\left\{1-\left(R_{F/C}\right)\right.\right.\right.$$

$$\left.\left(1-\left[1-R_{SW}R_R\right]^2\right)\right\}+3\left\{R_{F/C}\left(1-\left[1-R_{SW}R_R\right]^2\right)\right\}\left\{1-\left(R_{F/C}\right)\left(1-\left[1-R_{SW}R_R\right]^2\right)\right\}^2\Big]_{t_2}+$$

$$\left[3\left\{R_{F/C}\left(1-\left[1-R_{SW}R_R\right]^2\right)\right\}^2\left\{1-\left(R_{F/C}\right)\left(1-\left[1-R_{SW}R_R\right]^2\right)\right\}\right]\left[\left\{R_{F/C}\left(1-\left[1-R_{SW}R_R\right]^2\right)\right\}^2+\right.$$

$$\left.\left\{2R_{F/C}\left(1-\left[1-R_{SW}R_R\right]^2\right)\right\}\left\{1-\left(R_{F/C}\right)\left(1-\left[1-R_{SW}R_R\right]^2\right)\right\}\right]_{t_2}^{t_1}\left\{\left[R_{D\text{-}C\ BUS}\left(R_F^5+5R_F^4\left[1-R_F\right]+\right.\right.\right.$$

$$\left.\left.10R_F^3\left[1-R_F\right]^2\right)\right]_{t_1}^2\left[\left\{R_{D\text{-}C\ BUS}\left(R_F^5+5R_F^4\left[1-R_F\right]+10R_F^3\left[1-R_F\right]^2\right)\right\}^2+2\left\{\left(R_{D\text{-}C\ BUS}\right)\left(R_F^5+5R_F^4\left[1-R_F\right]\right.\right.\right.$$

$$\left.\left.\left.+10R_F^3\left[1-R_F\right]^2\right)\right\}\left\{1-\left(R_{D\text{-}C\ BUS}\right)\left(R_F^5+5R_F^4\left[1-R_F\right]+10R_F^3\left[1-R_F\right]^2\right)\right\}\right]_{t_2}\Big\}\left\{R_{\text{MAIN RELAY BUS}_{t_1}}\right\}$$

$$\left\{\left(R_{BATT}^3+3R_{BATT}^2\left[1-R_{BATT}\right]\right)_{t_1}\left(R_{BATT}^3+3R_{BATT}^2\left[1-R_{BATT}\right]\right)_{t_2}\right\}\left\{\left(1-\left[1-R_{BATT\ BUS\ A\&B}\right]^2\right)_{t_1}\left(1-\left[1-R_{BATT\ BUS\ A\&B}\right]^2\right)_{t_2}\right\}$$

$$\left\{\left(R_{BATT\ RELAY\ BUS}\right)_{t_1}\left(R_{BATT\ RELAY\ BUS}\right)_{t_2}\right\}\left\{\left(R_{CB}\right)\left(R_{SW}\right)\left(R_R\right)\left(R_{INV}\right)\left(1-\left[1-R_{SW}R_R\right]^2\right)\right]_{t_1}^3\left[\left\{\left(R_{CB}\right)\left(R_{SW}\right)\left(R_R\right)\left(R_{INV}\right)\right.\right.$$

$$\left.\left(1-\left[1-R_{SW}R_R\right]^2\right)\right\}^3+3\left\{\left(R_{CB}\right)\left(R_{SW}\right)\left(R_R\right)\left(R_{INV}\right)\left(1-\left[1-R_{SW}R_R\right]^2\right)\right\}^2\left\{1-\left(R_{CB}\right)\left(R_{SW}\right)\left(R_R\right)R_{INV}\right)\left(1-\left[1-R_{SW}R_R\right]^2\right)\Big\}$$

$$+3\left\{\left(R_{CB}\right)\left(R_{SW}\right)\left(R_R\right)\left(R_{INV}\right)\left(1-\left[1-R_{SW}R_R\right]^2\right)\right\}\left\{1-\left(R_{CB}\right)\left(R_{SW}\right)\left(R_R\right)\left(R_{INV}\right)\left(1-\left[1-R_{SW}R_R\right]^2\right)\right\}^2\Big]_{t_2}+$$

$$\left[3\left\{\left(R_{CB}\right)\left(R_{SW}\right)\left(R_R\right)\left(R_{INV}\right)\left(1-\left[1-R_{SW}R_R\right]^2\right)\right\}^2\left\{1-\left(R_{CB}\right)\left(R_{SW}\right)\left(R_R\right)\left(R_{INV}\right)\left(1-\left[1-R_{SW}R_R\right]^2\right)\right\}\right]_{t_1}\left[\left\{\left(R_{CB}\right)\left(R_{SW}\right)\right.\right.$$

$$\left.\left.\left(R_R\right)\left(R_{INV}\right)\left(1-\left[1-R_{SW}R_R\right]^2\right)\right\}^2+2\left\{\left(R_{CB}\right)\left(R_{SW}\right)\left(R_R\right)\left(R_{INV}\right)\left(1-\left[1-R_{SW}R_R\right]^2\right)\right\}\left\{\left(1-\left(R_{CB}\right)\left(R_{SW}\right)\left(R_R\right)\left(R_{INV}\right)\right.\right.\right.$$

$$\left.\left.\left.\left(1-\left[1-R_{SW}R_R\right]^2\right)\right\}\right]_{t_2}\Big\}\left\{\left[\left(R_{A\text{-}C\ BUS\ \#1\ \&\ \#2}\right)^2\right]_{t_1}\left[\left(R_{A\text{-}C\ BUS\ \#1\ \&\ \#2}\right)^2+2R_{A\text{-}C\ BUS\ \#1\ \&\ \#2}\left(1-R_{A\text{-}C\ BUS\ \#1\ \&\ \#2}\right)\right]_{t_2}\right\}$$

Figure 3-3. Reliability Electrical Power System Logic Diagram

Table 3-2. Electrical Power System Apportionment and Prediction

System and Component	Mission Success Reliability		Percent Unreliability = System Failure Rate / Total System Failure Rate	
	Apportioned	Predicted	Apportioned	Predicted
Fuel cell system (3)*	0.999553	0.994935	42.69	86.51
Fuel cell (1)	0.9718	0.90823		
Inverter System (3)**	0.999737	0.999258	25.12	12.67
Inverter (1)	0.9786	0.9643		
Battery System (3)*	0.999993	0.999993	0.67	0.12
Battery (1)	0.9984	0.9984		
Battery charger	0.995	0.9943		
D-C distribution	0.99968	0.999969	30.56	0.53
D-C bus	0.99999	0.99999		
A-C distribution	0.99999	0.999990	0.96	0.17
A-C bus	0.99999	0.99999		
TOTAL SYSTEM	0.998953	0.994149	100.00	100.00

*Denotes units in parallel redundancy
**Denotes units in stand-by redundancy

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Table 3-3. Fuel Cell Component Reliability Values

COMPONENT FAILURE-RATE ANALYSIS FOR APOLLO POWER CELL (INDEPENDENT MODULE)						
Component	Number (n) of Components per assy	Failure Rate (λ) Per 10^6 hr	Adjusted Failure Rate (λ_a) n. λ x 10^{-6} x K_T	Operating Time (hr)	λ_a x 10^{-6} x K_T x t	$R_{component}$
O_2 pressure regulator valve	1	4.25	9.50	400	3,800.0	0.99620
O_2 purge control valve	1	5.70	11.40	400	4,560.0	0.99544
H_2 pressure regulator valve	1	4.25	9.50	400	3,800.0	0.99620
H_2 bypass control valve	1	2.24	4.48	400	1,792.0	0.998208
H_2 pump and motor assembly	1	8.70	17.40	400	6,960.0	0.99304
H_2 regenerator	1	5.00	10.00	400	4,000.0	0.99600
H_2 condenser	1	5.00	10.00	400	4,000.0	0.99600
H_2 purge control valve	1	5.70	11.40	3	34.2	0.999997
*N_2 fill valve	1	2.80*	2.80	400	1,120.0	0.99890
N_2 control valve	1	1.9	2.80	400	1,120.0	0.99890
N_2 tank	1	1.0*	1.0	400	400.0	0.99960

Table 3-3. Fuel Cell Component Reliability Values (Continued)

COMPONENT FAILURE-RATE ANALYSIS FOR APOLLO POWER CELL (INDEPENDENT MODULE) (Continued)						
Component	Number (n) of Components per assy	Failure Rate (λ) Per 10^6 hr	Adjusted Failure Rate (λ_a) n . λ x 10^{-6} x K_T	Operating Time (hr)	λ_a x 10^{-6} x K_T x t	$R_{component}$
N_2 purge control valve	1	5.70	11.40	400	4,560.0	0.99544
N_2 pressure regulator	1	4.25	9.50	400	3,800.0	0.99620
Temperature sensor	3	3.3	19.8	400	7,920.0	0.99208
Module jacket	1	0.35*	0.35	400	140.0	0.99990
Thirty-one cell stack	1	116*	116	400	46,400.0	0.95466
Accumulator, glycol	1	6.18	12.36	400	4,944.0	0.995056
Pump and motor assembly, glycol	1	8.7	17.40	400	6,960.0	0.99304
Preheater O_2 and H_2	2	0.75	3.00	400	1,200.0	0.99880
Control valve, glycol	1	1.9	3.8	400	1,520.0	0.99898
Fill valve, glycol	1	2.80*	2.80	3	8.4	0.999992
Radiator	1	5.00	10.00	400	4,000.0	0.99600
Regenerator	1	5.00	10.00	400	4,000.0	0.99600
Bypass control valve, glycol	1	2.24	4.48	400	1,792.0	0.998208

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Table 3-3. Fuel Cell Component Reliability Values (Continued)

COMPONENT FAILURE-RATE ANALYSIS FOR APOLLO POWER CELL (INDEPENDENT MODULE) (Continued)						
Component	Number (n) of Components per assy	Failure Rate (λ) Per 10^6 hr	Adjusted Failure Rate $(\lambda_a) = n \cdot \lambda \times 10^{-6} \times K_T$	Operating Time (hr)	$\lambda_a \times 10^6 \times K_T \times t$	$R_{component}$
Shock mounts	3	0.35*	1.05	400	4,200.0	0.99580
Porous plugs	2	0.10	0.20	400	80.0	0.99992
Tubing and mechanical connectors	1	1.5*	1.5	400	600.0	0.99940
Wire and electrical terminals	1	1.2*	1.2	400	480.0	0.99952
Water/H_2 separator	1	2.5*	2.5	400	1,000.0	0.99900
Total Predicted Module Reliability = $\prod_{i=1}^{n} R_i$ =						0.90823
Legend: K_T - Environmental stress factor = 2 (constant)						
* Pratt & Whitney Aircraft reference						

test data will be used to provide assessments of achieved reliability. When compared to predicted reliability, the data will provide a measure of reliability growth.

FUEL CELL SYSTEM

Analysis

The fuel cell program has entered the intermediate phase of development testing. As a result, reliability effort at S&ID is being geared to re-evaluation of all existing reliability prediction studies based primarily on conceptual characteristics of previously designed power supplies and related state-of-the-art data.

This re-evaluation has shown that component failure-rate data submitted by the subcontractor in the early stages of product development does not reflect the anticipated failure rate of the equipment under consideration. The latest reliability studies conducted at S&ID were based on the most recent failure-rate data available.[1] Also considered were various application factors based on environmental conditions, performance rating, and life expectancy of the equipment. Table 3-3 gives the revised component failure rates for one of the three fuel cells in the system. Figures 3-4 and 3-5 display the Apollo fuel cell.

TEST PROGRAM

Qualification Test

Requirements of the qualification program were prepared for the EPS in SID 62-109-3 (General Test Plan, Ground Qualification Tests), describing the responsibilities of the suppliers and S&ID with regard to the qualification testing program.

A breakdown of the components to be used in the Apollo spacecraft was supplied in a table that included items to be tested, zone of location, criticality index, and the number of components to be tested.

A general qualification test schedule for the EPS and individual components also was given. Variations of the supplied schedule will be described in later reports.

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1 General failure rate values, presently used in systems analyses at S&ID, are obtained from Reliability Engineering Data Service Manual published by AVCO Corp., dated April 1962.

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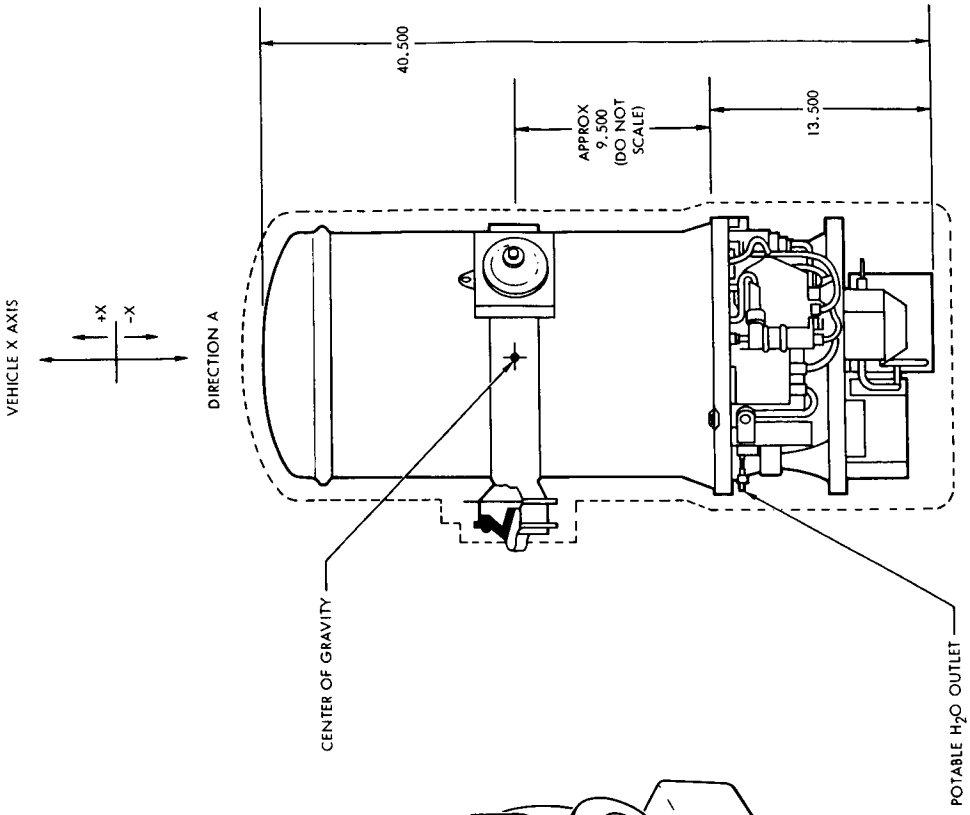

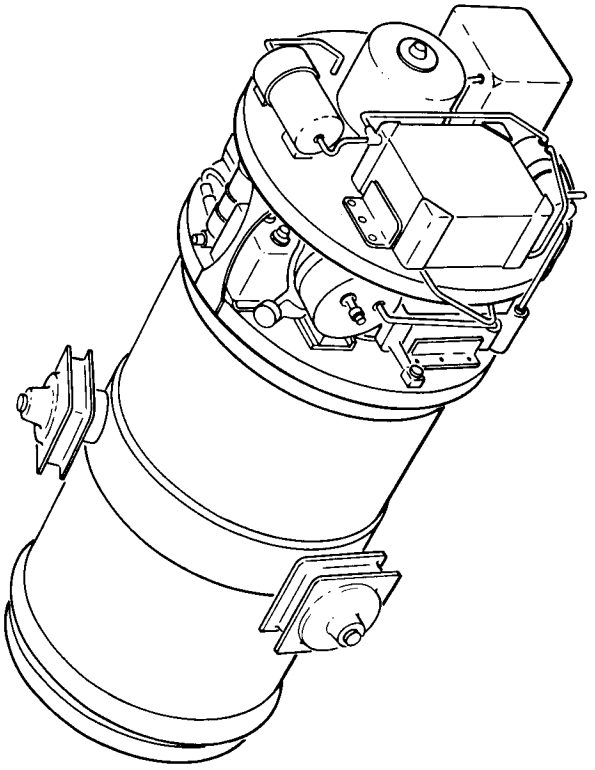

Figure 3-4. Fuel Cell Module

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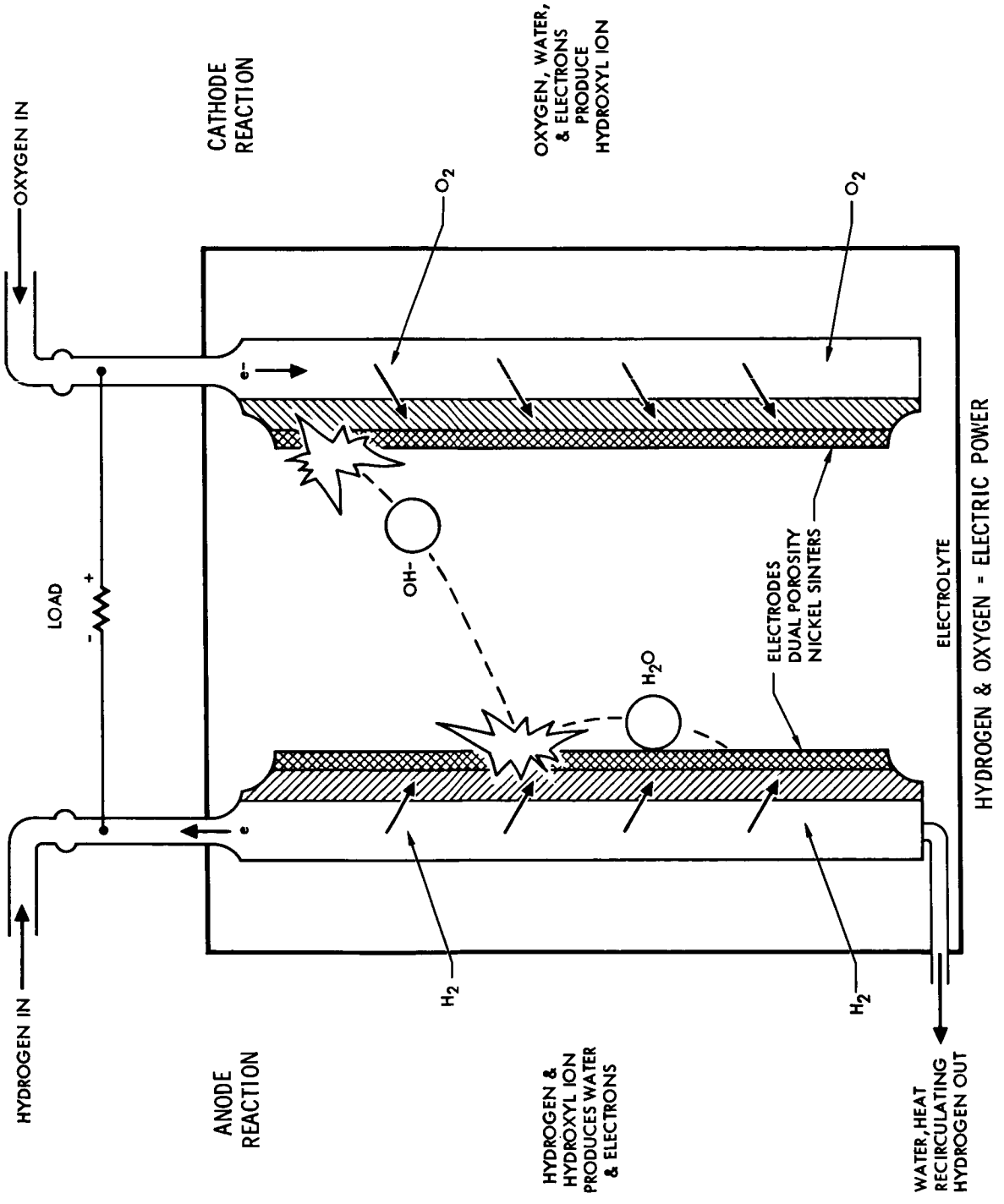

Figure 3-5. Basic Fuel Cell

General Test Program Status

All components are in the development phase. Development testing is scheduled to be completed in July 1964. The qualification program is not scheduled to start until October 1964.

Since the last quarter, the acceleration and acoustics tests were removed from off-limits testing.

Specific Development Tests

The first experimental power plant constructed to the bill of materials enabled a check of all mating dimensions and pinpointed design changes necessary to facilitate assembly. Operation of this powerplant provided output parameters for design studies and evaluation.

Roter-stater interference caused the initial two-bearing, pump-separater motor to stall. A third bearing was added to eliminate rotor cantilever action and, consequently, stalling. Because the vane was not sealing against the liner, a higher density vane was used to increase centrifugal force, and a relief slot was added to decrease vane-slot suction.

Initial tests on condensers, regenerators, and preheaters using nitrogen air and steam where required have resulted in some design modifications.

The position of the secondary regenerator was changed from horizontal to vertical because gravitational effects on the different densities of hot and cold fluids caused flow reversal. Also, the weight of the unit has been reduced by using 18 instead of 32 fins per inch. The lower fin density gives nearly optimum pressure drop distribution.

Loss of performance caused by maldistribution of flow has initiated redesign studies of the primary regenerator. Two changes have been rejected: one using a perforated tube and another using a fine wire cloth as flow spreaders in the upstream manifold. A larger internal surface area is under investigation.

Since axial conduction in the heat-exchanger material causes lowered performance, subscale models with different quantities of braze material are being tested. Reactant-preheater testing has just begun.

Variations in the output parameters from cell to cell are under investigation to bring about standardization. Single-cell vibration tests have resulted in structural failures at the inside diameter of the diaphragm on the oxygen side caused by natural frequencies of 13-21 cps in the KOM and

the container. Lower than expected, these frequencies will not allow the cells to withstand the present vibration requirement. This problem is under investigation. Insulation has been tested for heat transfer in a vacuum using flat samples.

Condenser

A subscale unit with special brazing was tested for variations caused by different quantitities of braze material in the core to determine the amount that will give maximum performance.

Glycol Loop Fill

A simulated glycol loop will be used to determine the best filling method. The loop is constructed of transparent tygon tubing.

Shock Mounts

Brillo-pad structure is being investigated for vacuum weld and weight loss at 300 F after 200 hours under 10^{-7} Torr pressure.

Seal Testing

Various configurations and compositions of Teflon material are under investigation.

Single Cell

Performance variation with temperature, electrolyte concentration, hydrogen humidity, reactant pressures, and electrode pressure differentials are being studied. Effects of impurities in the reactants, recovery time when purging with pure gas, and the relationship between hydrogen circulation rates and electrolyte concentration will be evaluated during the next quarter.

Subcontractor Activities

Subcontractor documentation reviewed during this period included the Reliability Program Plan, Quality Control Plan, Fuel Cell Program Plan, and the Development Test Plan. Comments concerning these documents have been forwarded to Pratt & Whitney Aircraft for action.

Primary development effort has been on fuel cell efficiency. Performance investigations conducted included (1) performance variation with

temperature and (2) electrolyte concentration, hydrogen humidity, reactant pressure, and differential electrode pressure. Additional studies are being conducted to determine the relationship between hydrogen circulation rates and electrolyte concentration. Also studied were cell response to various levels of impurities in the reactant supply and evaluation of recovery time when purging with pure gas.

Planned Activities

Re-evaluation of the fuel cell system will result in up-dating of current logic diagrams, math models, and failure-effects analyses. Reliability analyses and investigations will be performed on all system failures that occur during development tests. Acquired data will be used to assess reliability accomplishments and to determine and develop fuel cell reliability growth curves.

The following test activities are planned for the next quarter.

Fuel Cell Powerplant

An experimental powerplant will be tested for thermal balance and heat-loss in a vacuum chamber. An endurance run scheduled for 150 hours has completed 94 hours of operation. The run will be completed and evaluated next quarter.

The first PC3A-2 powerplant has been assembled and will be operated next quarter.

Primary Regenerator

A larger internal surface area subscale model with ruffled corrugation will be tested to see if it will give a more uniform distribution of flow.

Glycol Gear Pump

A vane pump will be designed and tested for backup. Tests will be performed with different gear sizes for correlation of sidewalls, clearance pumps, and leakage flow. Endurance runs will be started on 400-cycle pumps.

H_2 Pump Separator

To improve delivered flow, tests will be conducted to determine minimum power required and unit to unit variations.

Reactant Preheaters

Tests to be conducted with hydrogen, oxygen, and glycol for the first time on new test benches.

Flex Capsules

Flex capsules, to be used at headers and other correction points, will be subjected to vibration tests.

Insulation

Insulation in module-contour form will be tested during vacuum.

DISTRIBUTION SYSTEM

Analysis

Reliability analysis and evaluation of the distribution system has been limited because of the instability of basic design criteria. However, a preliminary analysis was performed on the system as presently established. The distribution system was divided into two sections: (1) AC distribution, which includes the AC busses, input and output circuit breakers, and all switches necessary for AC control and distribution (See Table 3-4 for AC power characteristics; Figure 3-6 is a schematic of the AC power distribution system) and (2) DC distribution, which includes the DC busses, battery bus, input and output circuit breakers, and all DC control and switching components (see Table 3-5 for DC power characteristics). The results of this analysis, given in Table 3-6, indicate that the present design will meet the required reliability goal. However, the existing design demonstrated the possibility of first-order failures that would be critical to both mission success and crew safety. To eliminate these conditions, a reliability crew safety design review was held, and necessary design action delegated.

Reliability/Crew Safety Design Review

An electrical power distribution reliability crew safety design review was conducted on 11 March 1963. Results indicated two significant problem areas:

1. Essential and non-essential telecommunications receive power from the same bus via a common switch, which is contrary to established practice and reliability concepts. The advisability of providing a separate switch for each equipment type was discussed, and Electrical Design was assigned the responsibility for remedial action.

Table 3-4. Apollo AC Power System Characteristics

Phases	Number	3
	Displacement	120 degrees
	Tolerance	± 2 degrees
Voltage	Nominal limits	115 to 200 volts
	Steady-state	115± 2 volts (average of 3 phases)
	Transient	115± 10 volts
	Unbalance	2 degrees (worst phase from average)
	Modulation	0.5 percent
Wave shape	Sine wave	
	Tolerance	
	Total distortion	5 percent
	Highest harmonic	4 percent
	Crest factor	1.414± 10 percent
Frequency	Nominal tolerance	400 cps
	Normal	Negligible (synch to master timing)
	Emergency	390± 10 cps (loss of master timing)
Rating	Volts-ampere	1250

Table 3-5. Apollo DC Power System Characteristics

- H_2 O_2 Fuel Cell Primary Power Source
- Fuel Cell Module Rating

  Power: 250-1500 Watts
  Voltage: 29± 2 Volts

- System Voltage at Load

  Nominal: 27.5 Volts
  Tolerance: ± 2.5 Volts
  Transient: 21 Volts - 1.0 Second
  (Fault) 32 Volts - 0.7 Second

- Minimum Weight Fuel Cell Requiring No System Voltage Regulating Equipment

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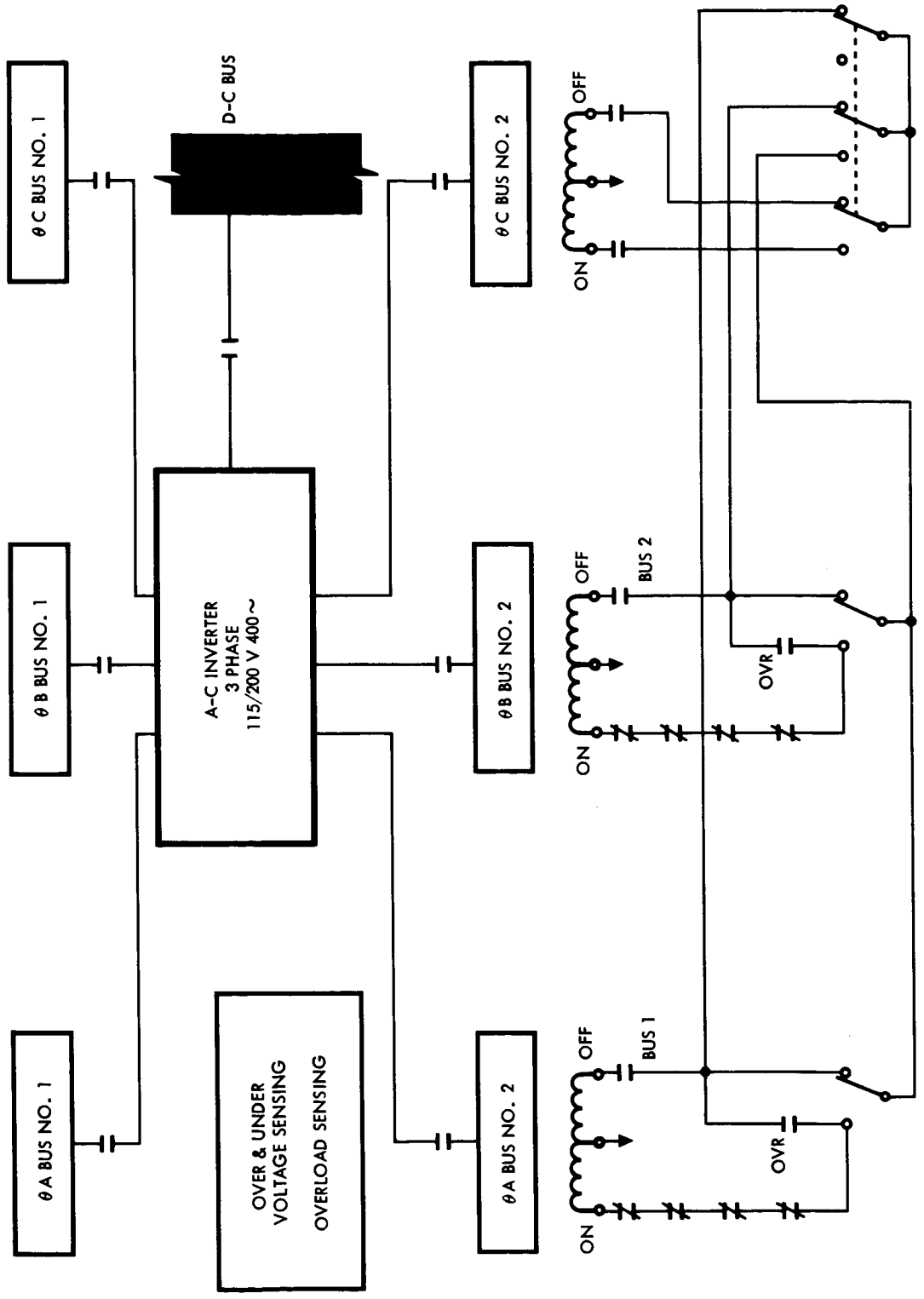

Figure 3-6. A-C Power

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Table 3-6. Electrical Power System Apportionment and Prediction

Electrical Power Prediction and Apportionment Distribution System		
	Mission Success Reliability	
System	Apportioned	Predicted*
AC distribution	0.99999	0.999990
DC distribution	0.99968	0.999969
Distribution system	0.99967	0.999959
*Based on state-of-the-art failure rates		

Distribution System Component State-of-the-Art Failure Rates	
Component	Failure Rate/10^6 Hr
Buses (ac, dc, battery, main relay)	0.02976
Switch	2.70
Relay	2.50
Fuse	1.00
Circuit breaker	2.00

2. No power is available on the AC busses during the switching interval required in the event of an inverter failure. Since only one of the three inverters in the system is operating at any particular time, a failure would have to be observed by the crew, and manual switching accomplished. During this interval, the suit compressors would be inoperative, thus possibly endangering the crew. To prevent loss of power, an automatic sensing and switching feature with manual override was proposed for incorporation into the system. Reliability will review NASA criteria for equipment failure versus crew capability to cope with failures. Life Sciences and Electrical Design will follow-up thereafter.

Table 3-7 shows the action items discussed during the design review meeting. Areas of reliability and present status are given on each item.

Table 3-7. Electrical Distribution System Design Review Action Items

Action Item	Description of Task	Action By	Status
1. Loss of the No. 1 fuel cell neutral return would leave the other two connected fuel cells, and hence the circulator and separator motors, without a neutral return. Should these motors have individual neutral return?	Verify by testing whether individual neutral returns are required and redesign if necessary.	EPS Design Group	Tests are in progress.
2. Essential telecommunications and non-essential telecommunications receive power from the same bus through a common switch. Should essential and non-essential functions be separated? Should another switch be added to provide separate power sources?	Redesign so that individual switches separate essential and non-essential functions.	EPS Design Group	A study is being made.
3. In some instances, equipment is attached through one double throw switch to both the A and B main buses. Redundant power sources are thus provided. This advantage is somewhat nullified, however, by using a common	Determine the probability of this crew error occuring. Further study was indicated.	Reliability and Life Sciences	No action to date.

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Table 3-7. Electrical Distribution System Design Review Action Items (Continued)

Action Item	Description of Task	Action By	Status
switch. Should separate switches be used? The design group felt that through crew error separate switches could lead to both buses being connected to the same load.			
4. A failure in the inverter must be observed by the crew. Then manual switching to another inverter must be accomplished. During this interval, there is no power on the a-c busses. Should an automatic sensing and switching feature with manual override be substituted for the manual switch? During the interval noted above the suit compressors could be inoperative. Would this be critical to the crew?	A study is required to determine the times involved, to review NASA's criterion for equipment failure versus crew capability, and to determine if an automatic sensing and switching feature with an automatic override is indicated.	Reliability, Life Sciences, and EPS Design Groups	A study is in progress.
5. Should inverters No. 1 and 2 have the same switching capabilities as No. 3? The design group stated that the probability of losing two in-	A study to determine the degree of remoteness is to be made.	Reliability	No action to date.

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Table 3-7. Electrical Distribution System Design Review Action Items (Continued)

Action Item	Description of Task	Action By	Status
verters or either the A or B bus was remote.			
6. Has a trade-off study been made to compare the above hermetically sealed switches for Controls & Displays with an environment-sealed type switch? The Displays and Controls Group stated that an environment-sealed switch was under study, but that a trade-off versus the hermetically-sealed type had not been made.	Perform a trade-off study.	EPS Design Group and Reliability Component Technology	A plan of action has been prepared and is in the process of implementation.
7. When the cutter severs the wire bundles between the C M and S M during separation, are the lines in the bundle shorted out? What provisions will be made for protecting the battery circuits if this occurs? A manual switching feature in incorporated in the C M to protect the battery circuits.	Prepare a study of the desirability of making this feature automatic, with a manual override.	Reliability	No action to date.

Special Studies

In response to a NASA request, Reliability Engineering supported a design study for command module-to-service module electrical interconnections. The designs analyzed were as follows:

1. Dual umbilical connectors through the heat shield, with redundant pyro circuits individually shielded and a single separation mechanism for both connectors.

2. Quad umbilical connectors through the heat shield - two for redundant electrical and r.f. circuits and two for redundant pyro circuits - and a single disconnect mechanism for each dual set.

3. A single-ribbon cable bonded to the outside of the spacecraft. Separation would be effected by shaped charge, and pyro circuits would be isolated by grouping with both individual and group shields, all circuits redundant. Isolation of pyro circuits is maintained at the service-module end of the cable by use of separate connectors for electrical RF cables.

4. Dual ribbon cables bonded to the outside of the spacecraft. Separation of each bonded cable would be by shaped charge, without redundancy.

A preliminary reliability analysis was performed for each of the above design assumptions. A summary of the results of these analyses given in Table 3-8 indicates that the single-ribbon cable approach is superior from a reliability standpoint.

Table 3-8. Distribution System Predicted Reliability Values

Design	Predicted Reliability	Predicted Failures per Million Missions
1	0.999980	20
2	0.999963	37
3	0.999998	2
4	0.999994	6

Design 4 was approximately 8 to 10 pounds heavier than design 3; therefore, Reliability Engineering recommended that effort be directed toward design 3 (single-ribbon cable).

Planned Activities

During the next report period, tests will be continued on distribution system hardware. Information derived from these tests will be analyzed and incorporated into the system math model for periodic updating.

BATTERY SYSTEM

Analysis

During this report period, organization was continued of reliability requirements relative to the battery development program. A reliability program was prepared containing a general analysis of the design concept of the battery and the procedures to be used in establishing the environmental withstanding and capability limits of the components and prototype battery assemblies. These procedures, when applied in conjunction with preliminary test data, will be used to determine reliability growth. Figure 3-7 is a schematic of the battery system. Figure 3-8 is a physical picture of the battery.

Special Studies

At the request of Design Engineering, a three-step reliability analysis was performed on a proposed battery venting system as follows:

1. Failure mode analysis on the proposed system
2. Reliability design evaluation
3. Suggested design improvement

Table 3-9 gives a failure mode analysis of the proposed design, indicating the type of failure and possible effects on system performance. The results of this analysis were submitted to Design for consideration in determining the final design approach.

Test Program

Preliminary testing efforts concerned with developing assurance against component failure caused by environmental conditions included the vibration testing of performance-tested pre-prototype battery cells at various

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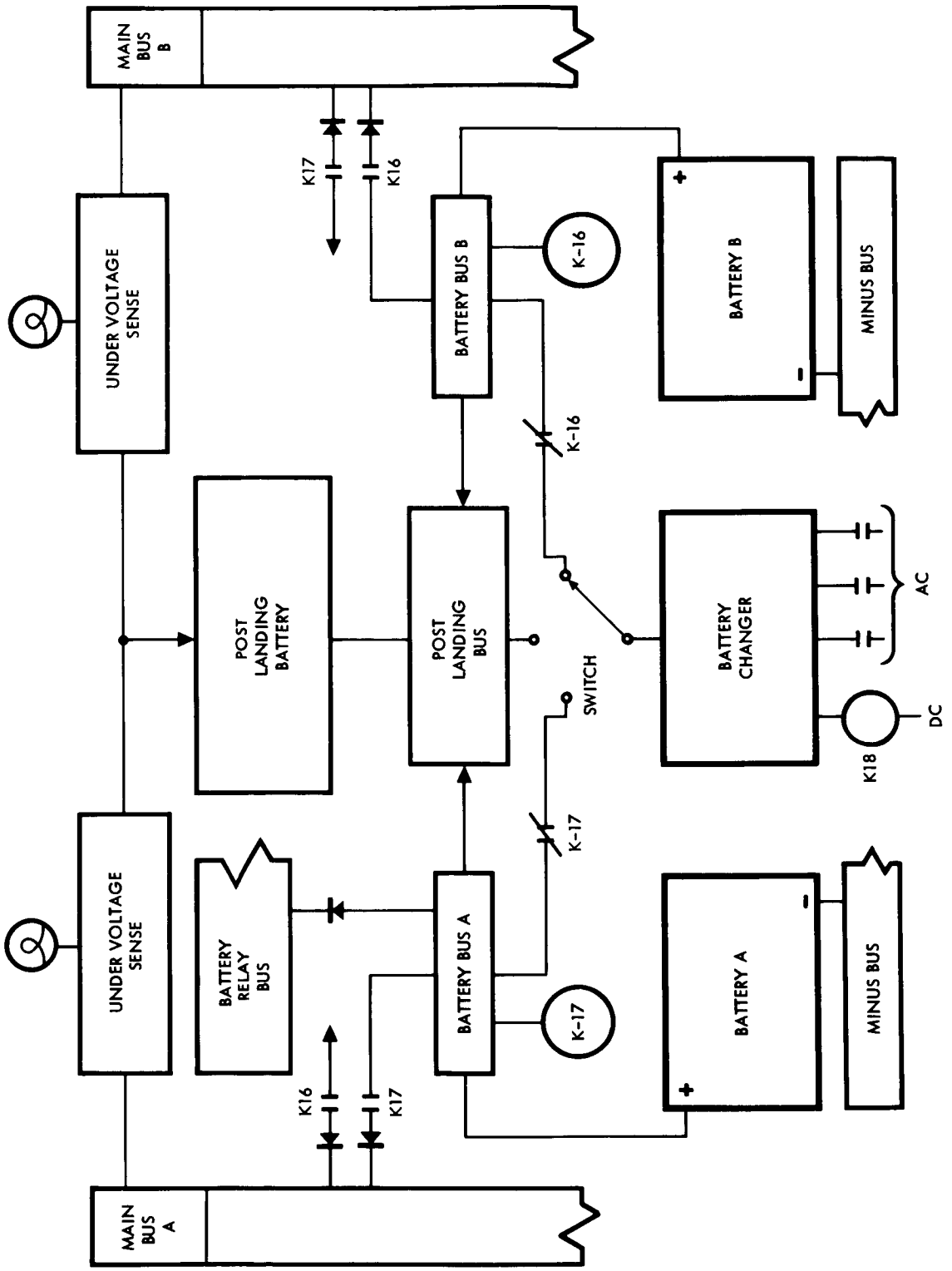

Figure 3-7. Battery System

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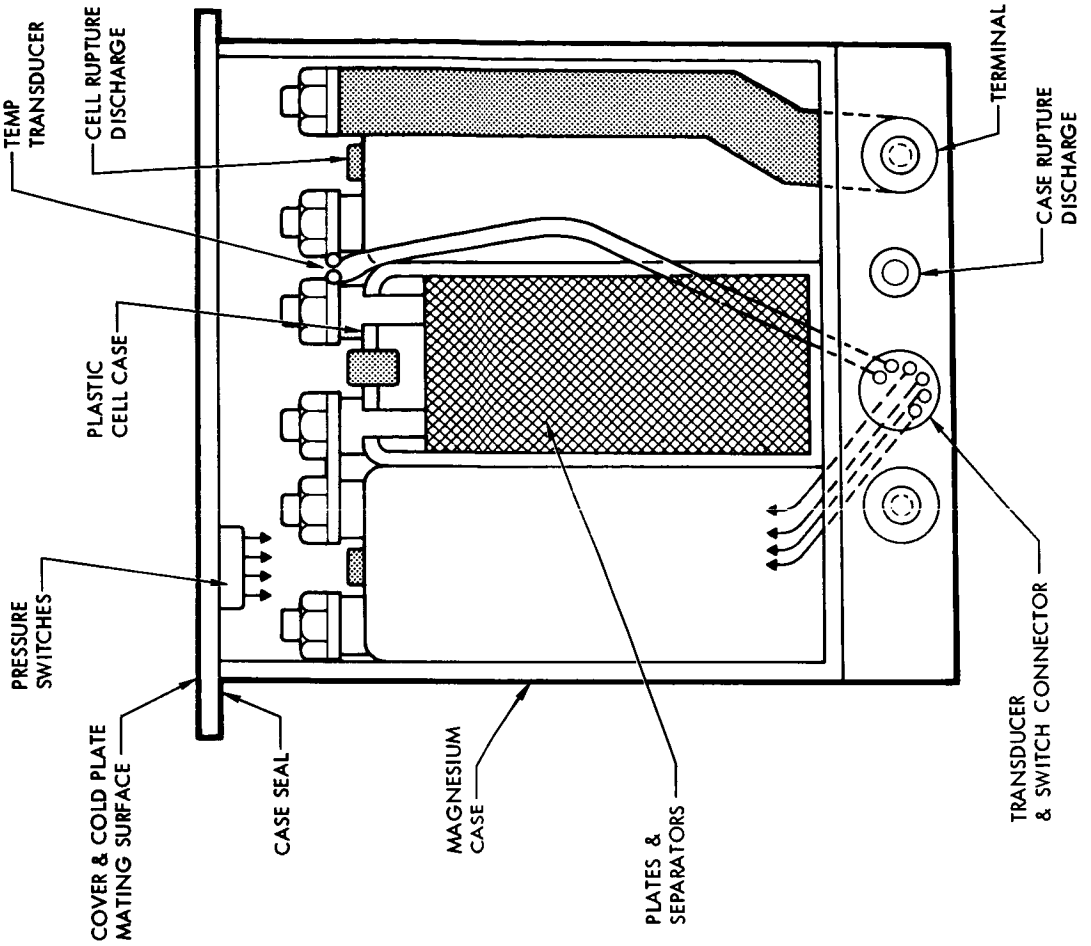

CAPACITY	25 AH
WEIGHT	
SIZE	
RELIABILITY	0.9984
VOLTAGE	29±2 VOLTS
STATE RECEIVED	WET DISCHANGE
SHELF LIFE	1 YEAR
CYCLE LIFE	6 CYCLES

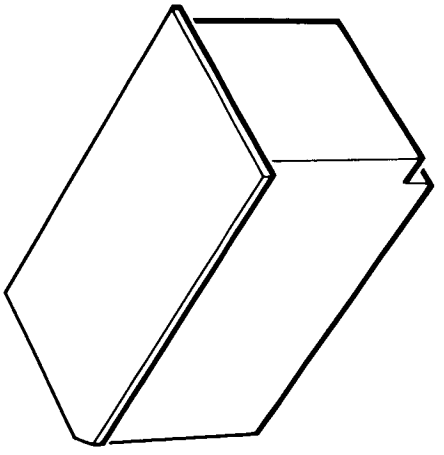

Figure 3-8. Proposed Battery Design

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Table 3-9. Failure Mode Analysis

Component	Failure Mode	Remarks
Rupture disc	Leak	Pressure buildup between disc and relief valve would alter burst pressure.
	Fragmentation upon burst	Possible contamination of relief valve is sufficient to prevent positive reseat.
	Failure to burst	Possible battery case rupture.
Relief valve	Failure to open	Increase in back pressure on remaining discs, resulting in back burst.
	Failure to reseat	No apparent effect providing no prior system failure has occurred that will open cabin to space.
Tubing and connections	Leak	Possible contamination of crew compartment.
Cell relief valve	Leak	Slow buildup of pressure within battery case.
	Failure to relieve	Cell rupture.
	Fail open	Loss of electrolyte. Slight degredation of capacity.

vibration levels equal to and exceeding the specified environmental requirements. Evidence indicates that the present design is adequate to meet present specified vibration requirement levels and that the cells are capable of withstanding up to twice the specification levels. Their capability will be further investigated in subsequent tests.

The adequacy of normal battery performance and cycle life is being determined by analyzing the life cycle performance characteristics of cells subjected to partial life test. No definite statement about the reliability of required cycle life is possible yet.

Planned Activities

During the next report period, primary reliability effort will be directed toward updating the battery failure mode and effects analysis, logic network, and reliability prediction. Special reliability studies will be directed toward establishing cycle life performance capability of the battery.

BATTERY CHARGER

Analysis

Reliability effort on the battery charger has progressed well during the period covered by this report. Initial predictions indicate an inherent reliability of 0.9943, which is only slightly below the apportioned value of 0.9950. The apportioned value may be attained by incorporation of high-reliability components.

Special Studies

A listing of battery charger components has been submitted to S&ID by the supplier, and a study is in progress to determine validity of the failure rates used. Table 3-10 is a breakdown of components and failure rates used in the current battery charger circuitry. This table indicates also the percentage of unreliability contributed by various types of components, thus indicating areas for primary reliability effort. As analysis is completed, coordination meetings with the supplier will be held to implement the necessary tradeoffs to assure that reliability requirements are met.

Subcontractor Activities

International Telephone and Telegraph has completed the design of two breadboard circuits for the battery charger.

Planned Activities

During the next report period, major reliability effort will be directed toward establishing a math model and logic diagram for use in predicting reliability growth. Also, parts will be reviewed to insure that only high-reliability components are selected.

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Table 3-10. Battery Charger Component Reliability Values

Components	Quantity	Total Failure Rate (λ_c) (Percent/1000 hr)	Percent Total Failure Rate $\left(\frac{\lambda_c}{2.869}\right)$
Transistors	8	0.054	1.88
Diodes	19	0.096	3.35
Resistors	24	0.346	12.05
Capacitors	14	0.825	28.75
Inductors	8	0.720	25.09
Transformers	3	0.468	16.38
Connector	1	0.360	12.05
TOTAL		2.869	100.00

$$MTBF = \int_0^\infty R\,dt = \int_0^\infty e^{-\lambda t}\,dt = 1/\lambda$$

$$= \frac{1}{2.869 \times 10^{-5}} = 34,855$$

$$R = e^{-\lambda t} = e^{-(2.869 \times 10^{-5})(200)}$$

$$R = 0.9943$$

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INVERTER SYSTEM

Analysis

During this quarter, reliability effort has been directed toward selection of high-reliability parts and components used in the inverter. Analyses indicate an increase in inverter state-of-the-art reliability from 0.9638 to 0.9643.

Problem Areas

Reliability analyses to date indicate that the present design is incapable of meeting the apportioned reliability of 0.9786. Utilizing the latest and most accurate failure data, the present reliability of the inverter is predicted to be 0.9643, which represents a failure rate of 9.08 percent per 1000 hours. (See Table 3-11.) The apportioned failure rate is 5.4 percent per 1000 hours. To attain this value, a 40.5 percent reduction of the present failure rate is mandatory.

Reduction of the present failure rate can be accomplished by means of a component improvement program or by utilization of redundant components. The effects and accomplishments of a parts improvement program are costly and require excessive time, and in many cases programs of this nature have been ineffective. Therefore, Apollo Reliability Engineering does not recommend this approach.

Design analysis of the inverter circuitry has indicated that the required failure-rate reduction may be attained by utilizing high-reliability components and incorporating redundancy where possible. Westinghouse is using metal film resistors manufactured by the Daven Resistor Company. NAA/S&ID Reliability has recommended that these resistors be replaced by the minuteman high-reliability XLT resistors manufactured by the International Resistance Company. Table 3-11 indicates the advantages of this replacement. To further decrease the failure rate, it is also recommended that in some diode applications, replacement of the single diode by a quad arrangement be incorporated where possible. The present design contains 120 diodes of which 80 contribute 34.5 percent of the present failure rate. If these 80 diodes are replaced with quad groups, the inverter failure rate will be reduced as indicated in Table 3-12.

It should be noted that weight has been considered in this tradeoff study, and no penalty resulted.

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Subcontractor Activities

Westinghouse Electric Corporation has completed the basic design for the static inverter. An operating breadboard model has been completed and was demonstrated to NAA personnel in Lima, Ohio.

Table 3-11. Single Inverter Component Reliability Values

Components	Quantity	Total Failure Rate (λ_c) (Percent/hr)	Percent Total Failure Rate $\left(\frac{\lambda_c}{9.08}\right)$
Transistors	55	1.856	20.4
Diodes	120	3.223	35.6
Resistors	99	0.9627	10.6
Capacitors	35	0.2083	2.3
Inductive	32	2.70	29.7
Terminals	13	0.13	1.4
TOTAL		9.08	100.0

$$MTBF = \int_0^\infty Rdt = \int_0^\infty e^{-\lambda t}\, dt = \frac{1}{\lambda}$$

$$= \frac{1}{9.08X10^{-5}} = 11,013 \text{ hr}$$

$$R = e^{-\lambda t} = e^{-(9.08X10^{-5})(400)}$$

$$= 0.9643$$

Planned Activities

During the next quarter, reliability effort will be directed toward updating the failure-effects analysis. Logic diagrams and math models will be utilized to predict significant changes in state-of-the-art reliability levels as design improvement and test data are analyzed.

[redacted]

Table 3-12. Inverter Failure Rate

Components	Total Weight (oz)	ΔWeight (oz)	Failure Rate Contributed by Components	Percent of Present Failure Rate
Resistors				
64 Daven	3.37	-2.5571	0.64%/1000 hr	-6
64 XLT	0.8129		0.10%/1000 hr	
Diodes				
80	0.68	+2.04	3.13%/1000 hr	-34.5
80 quad gps	2.72		0.01%/1000 hr	
Net Charge		-0.517		-40.5

[redacted]

CRYOGENIC STORAGE

SUMMARY

As a result of simplifying the cryogenic storage system (CSS), numerical reliability analyses now indicate that the system is capable of meeting the apportioned mission success and crew safety goals employing state-of-the-art components. The revised predicted cryogenic storage system mission success and crew safety numerical reliability values versus apportioned values are summarized in Table 3-13.

Table 3-13. Cryogenic Storage System Apportionment and Prediction

System	Mission Success Reliability		Crew Safety Reliability	
	Apportioned	Predicted*	Apportioned	Predicted*
Cryogenic Storage System	0.998900	0.999149	0.999990	0.999999
*Based on State-of-the-Art Failure Rates				

The predicted values of mission success and crew safety applicable to the cryogenic storage system, reported in the Fourth Quarterly Reliability Status Report (SID 62-557-4), page 67, were 0.9946 and 0.99977, respectively.

ANALYSIS

The CSS as originally conceived and illustrated in Figure 9 of the Fourth Quarterly Reliability Report (SID 62-557-4) was extremely complex. A vigorous analysis of this complex system indicated that several of the components provided for alternate modes of operation had critical failure rates greater than those of the component for which they provided back-up.

It was apparent to both the subcontractor, Beech Aircraft Corp., and NAA/S&ID that a redesign of the system was required, if apportioned mission success and crew safety goals were to be met. As a result, a simplified system has been established that eliminates many of the components in the original system and has a greatly increased probability of mission success and crew safety reliability. The revised CSS design has been accepted by S&ID and is schematically illustrated in Figure 3-9. Figure 3-10 shows

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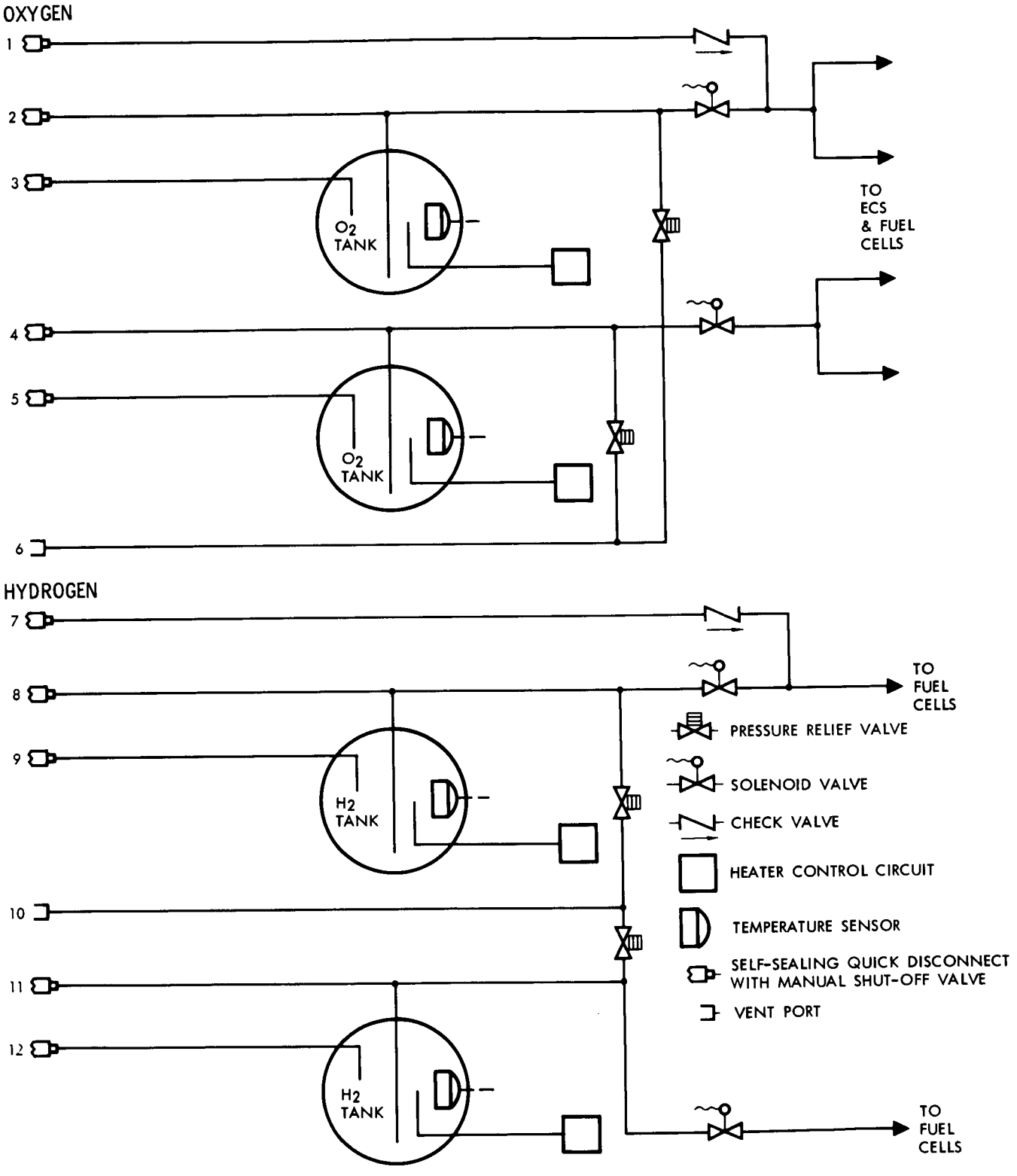

Figure 3-9. Cryogenic Storage System Schematic Diagram

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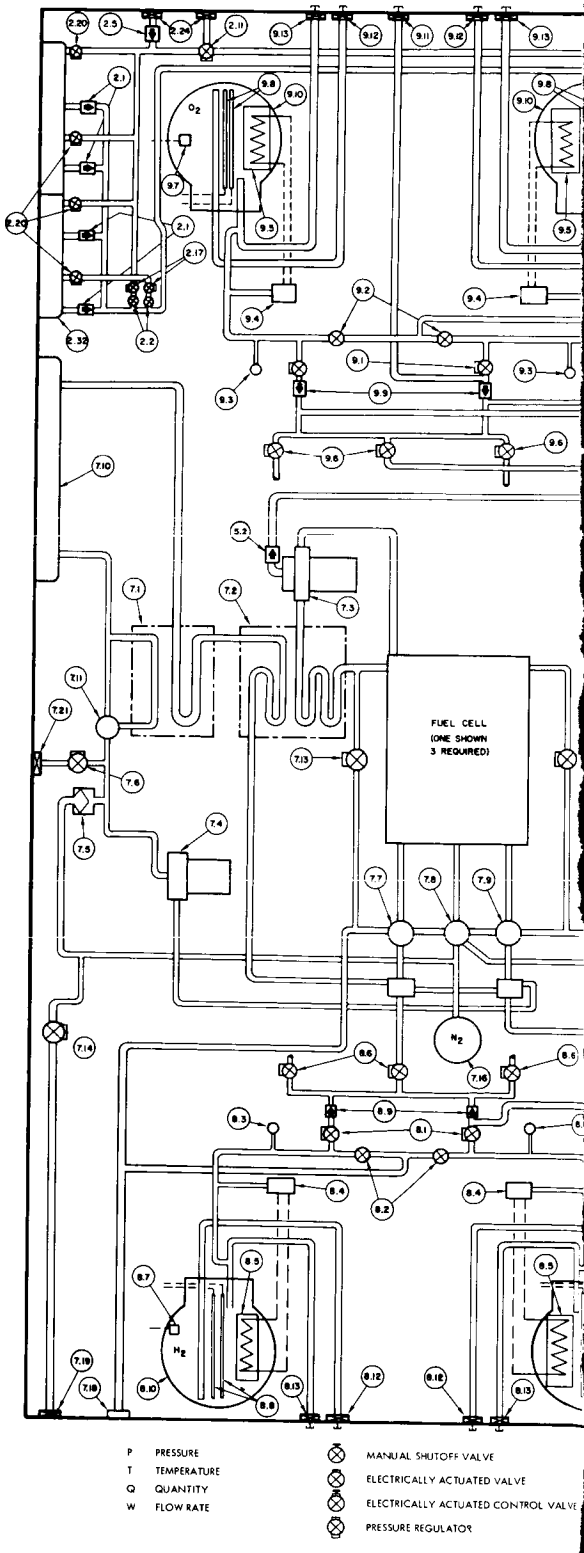
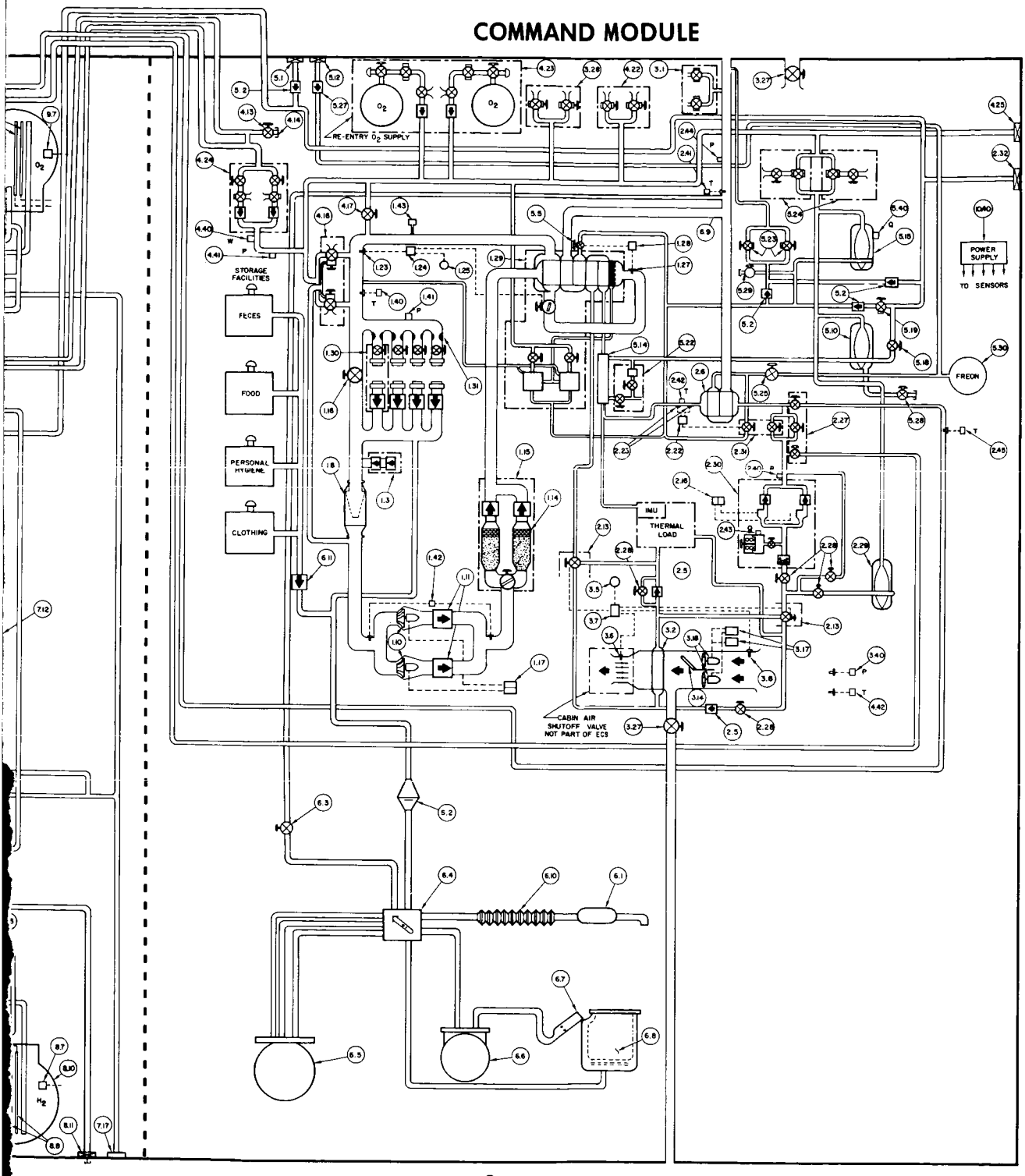
SERVICE MODULE



COMMAND MODULE



PRESSURE REGULATOR AND RELIEF VALVE
PRESSURE RELIEF VALVE
PRESSURE RELIEF VALVE WITH MANUAL OVERRIDE
ELECTRICALLY ACTUATED 3-WAY VALVE
ELECTRICALLY ACTUATED 3-WAY VALVE WITH MANUAL OVERRIDE
CHECK VALVE
SELF SEALING QUICK DISCONNECT
QUICK DISCONNECT
SELF SEALING QUICK DISCONNECT WITH MANUAL CLOSURE

Fi

ITEM NO.	NO. REQ.	DESCRIPTION
		GROUP I - PRESSURE SUIT CIRCUIT
1.3	1	SUIT CIRCUIT RETURN AIR CHECK VALVE ASSEMBLY
1.8	1	DEBRIS TRAP
1.10	2	SUIT COMPRESSOR
1.11	2	SUIT COMPRESSOR CHECK VALVE
1.14	28	CO_2 AND ODOR ABSORBER REMOVABLE CONTAINER (26 REPLACEMENTS)
1.15	1	CO_2 AND ODOR ABSORBER PACKAGE
		$LiOH$ CANISTER (2)
		CHECK VALVE (2)
		ISOLATION VALVE
1.16	1	SUIT BYPASS VALVE
1.17	2	SUIT COMPRESSOR SELECTOR SWITCH
1.23	1	SUIT AIR TEMPERATURE SENSOR
1.24	1	SUIT AIR TEMPERATURE CONTROL
1.25	1	SUIT AIR TEMPERATURE SELECTOR
1.27	1	SUIT EVAPORATOR TEMPERATURE SENSOR
1.28	1	SUIT EVAPORATOR TEMPERATURE CONTROL
1.29	1	HEAT EXCHANGER PACKAGE
		REGENERATIVE HEAT EXCHANGER
		REGENERATIVE HEAT EXCHANGER BYPASS VALVE
		SUIT EVAPORATOR
		GLYCOL-TO-SUIT AIR HEAT EXCHANGER
		WATER SEPARATOR
		WATER SEPARATOR PUMP ASSEMBLY
1.30	4	SUIT HOSE CONNECTOR PACKAGE
		SUIT SHUTOFF VALVE
		SUIT SUPPLY CONNECTION
		SUIT RETURN
		CHECK VALVE
1.31	4	SUIT FLOW LIMITER
		GROUP II - WATER - GLYCOL CIRCUIT
2.1	4	SPACE RADIATOR OUTLET CHECK VALVE
2.2	2	GLYCOL PRESSURE RELIEF VALVE
2.5	3	GLYCOL CHECK VALVE
2.6	1	GLYCOL EVAPORATOR
2.11	1	GLYCOL SHUTOFF VALVE
2.13	2	CABIN TEMPERATURE CONTROL VALVE
2.16	2	GLYCOL PUMP SELECTOR SWITCH
2.17	2	GLYCOL SHUTOFF VALVE
2.20	4	SPACE RADIATOR ISOLATION & VENT VALVE
2.22	1	GLYCOL TEMPERATURE CONTROL
2.23	2	GLYCOL TEMPERATURE SENSOR
2.24	2	GLYCOL FILL & VENT CONNECTION
2.27	1	MANUAL SHUTOFF VALVE
2.28	5	GLYCOL SHUTOFF VALVE
2.29	1	GLYCOL RESERVOIR
2.30	1	GLYCOL PUMP ASSEMBLY
		ACCUMULATOR
		ISOLATION VALVE
		PUMP (2)
		CHECK VALVE (2)
		FILTER
2.31	1	GLYCOL TEMPERATURE CONTROL VALVE
2.32	4	SPACE RADIATOR
2.33	1	LEM WATER SUPPLY QUICK DISCONNECT
		GROUP III - COMMAND MODULE PRESS & TEMP. CONTROL SYSTEM
3.1	1	CABIN OUTFLOW, PRESS. REG. & NEGATIVE RELIEF VALVE
3.2	1	CABIN HEAT EXCHANGER
3.5	1	CABIN TEMPERATURE SELECTOR
3.6	1	CABIN TEMPERATURE ANTICIPATOR
3.7	1	CABIN TEMPERATURE CONTROL
3.8	1	CABIN TEMPERATURE SENSOR
3.14	1	CABIN BLOWER DIVERTER VALVE
3.17	2	CABIN BLOWER SELECTOR SWITCH
3.18	2	CABIN RECIRCULATING BLOWER
3.27	2	CABIN VENT VALVE
3.28	1	CABIN PRESSURE REGULATOR
		GROUP IV - OXYGEN SUPPLY
4.13	1	BACK PACK SUPPLY SHUTOFF & PRESSURE RELIEF VALVE
4.14	1	BACK PACK SUPPLY CAP
4.16	1	DEMAND PRESSURE REGULATOR & RELIEF VALVE
4.17	1	MANUAL O_2 SUPPLY VALVE
4.22	1	EMERGENCY O_2 INFLOW CONTROL VALVE
4.23	1	RE-ENTRY O_2 SUPPLY SYSTEM
		O_2 TANK (2)
		O_2 FILL PORT (2)
		VALVE ASSEMBLY (2)
		PRESSURE REGULATOR (2)
		RELIEF VALVE (2)
		CHECK VALVE (2)
4.24	1	OXYGEN PRESSURE REGULATOR ASSEMBLY
		SHUTOFF VALVE (2)
		PRESSURE REGULATOR (2)
		RELIEF VALVE (2)
		CHECK VALVE (2)
4.25	1	LEM OXYGEN SUPPLY QUICK DISCONNECT
		GROUP V - WATER SUPPLY SYSTEM
5.1	1	WATER SUPPLY QUICK DISCONNECT
5.2	5	WATER CHECK VALVE
5.5	1	SUIT EVAPORATOR WATER INFLOW CONTROL VALVE
5.10	1	POTABLE WATER TANK
5.12	1	FREON QUICK DISCONNECT VALVE
5.14	1	WATER CHILLER
5.15	1	WASTE WATER TANK
5.18	1	WATER SHUTOFF VALVE
5.19	1	WATER TANK PRESSURE RELIEF VALVE
5.22	1	POTABLE WATER SUPPLY ASSEMBLY
		SUPPLY VALVE (2)
		HEATER
5.23	2	WATER PRESSURE RELIEF VALVE
6.24	2	TANK PRESSURE CONTROL & RELIEF VALVE
5.25	1	FREON FLOW VALVE
5.27	1	FREON CHECK VALVE
5.28	1	SURVIVAL WATER SHUTOFF VALVE
5.29	1	EMERGENCY WATER SHUTOFF VALVE
5.30	1	FREON RESERVOIR TANK

ITEM NO.	NO. REQ	DESCRIPTION
		GROUP VI - WASTE MANAGEMENT SYSTEM
6.1	1	VACUUM BAG
6.2	1	BLOWER
6.3	1	SHUTOFF VALVE
6.4	1	SELECTOR SWITCH
6.5	1	URINE SEPARATOR
6.6	1	CHEMICAL TANK
6.7	1	RELIEF TUBE
6.8	1	FECES BAG
6.9	1	URINE OVERBOARD
6.10	1	FLEXABLE HOSE
6.11	1	STORAGE FACILITIES CHECK VALVE
		GROUP VII - FUEL CELL SYSTEM
7.1	1	REGENERATOR
7.2	1	CONDENSOR
7.3	1	CIRCULATION SYSTEM
		PUMP
		SEPARATOR
		MOTOR
7.4	1	SECONDARY LOOP COOLANT PUMP
7.5	1	ACCUMULATOR AND PRESSURE REGULATOR
7.6	1	FILL VALVE
7.7	1	HYDROGEN PRESSURE REGULATOR
7.8	1	NITROGEN PRESSURE REGULATOR
7.9	1	OXYGEN PRESSURE REGULATOR
7.10	1	FUEL CELL RADIATOR
7.11	1	TEMPERATURE CONTROL REGULATOR & REGENERATOR BYPASS VALVE
7.12	1	OXYGEN PURGE SYSTEM VALVE
7.13	1	HYDROGEN AUXILIARY COOLING SYSTEM VALVE
7.14	1	NITROGEN TANK FILL VALVE
7.16	1	NITROGEN TANK
7.17	1	OVERBOARD VENT
7.18	1	HYDROGEN VENT
7.19	1	NITROGEN SUPPLY QUICK DISCONNECT
7.21	1	COOLANT SUPPLY QUICK DISCONNECT
		GROUP VIII - HYDROGEN SYSTEM
8.1	2	TANK SHUT-OFF VALVE
8.2	2	PRESSURE RELIEF VALVE
8.3	2	PRESSURE TRANSDUCER
8.4	2	PRESSURE SWITCH
8.5	2	ELECTRICAL HEATER
8.6	3	FUEL CELL SUPPLY SHUTOFF VALVE
8.7	2	TEMPERATURE SENSOR PROBE
8.8	2	QUANTITY SENSOR PROBE
8.9	2	CHECK VALVE
8.10	2	HYDROGEN TANK
8.11	1	HYDROGEN PURGE LINE
8.12	2	HYDROGEN FILL LINE
8.13	2	HYDROGEN FILL VENT
		GROUP IX - OXYGEN SYSTEM
9.1	2	TANK SHUTOFF VALVE
9.2	2	PRESSURE RELIEF VALVE
9.3	2	PRESSURE TRANSDUCER
9.4	2	PRESSURE SWITCH
9.5	2	ELECTRICAL HEATER
9.6	3	FUEL CELL SUPPLY SHUTOFF VALVE
9.7	2	TEMPERATURE SENSOR PROBE
9.8	2	QUANTITY SENSOR PROBE
9.9	2	CHECK VALVE
9.10	2	OXYGEN TANK
9.11	1	OXYGEN PURGE LINE
9.12	2	OXYGEN FILL LINE
9.13	2	OXYGEN FILL VENT
		GROUP I - PRESSURE SUIT INSTRUMENTATION
1.40	1	TRANSDUCER, TEMP., INLET, SUIT SUPPLY MANIFOLD
	1	SENSOR
	1	AMPLIFIER
1.41	1	TRANSDUCER, PRESS., INLET SUIT SUPPLY MANIFOLD
1.42	1	TRANSDUCER, DIFFERENTIAL PRESS., SUIT COMPRESSOR
1.43	1	TRANSDUCER, PARTIAL PRESSURE, CO_2
		GROUP II - WATER - GLYCOL CIRCUIT INSTRUMENTATION
2.40	1	TRANSDUCER, PRESS., OUTLET, GLYCOL PUMP
2.41	1	TRANSDUCER, TEMP., GASEOUS, OUTLET, GLYCOL EVAPORATOR
	1	SENSOR
	1	AMPLIFIER
2.42	1	TRANSDUCER, TEMP., LIQUID, OUTLET, GLYCOL EVAPORATOR
	1	SENSOR
	1	AMPLIFIER
2.43	1	TRANSDUCER, PRESS., QUANTITY MEASUREMENT, GLYCOL ACCUMULATOR
2.44	1	TRANSDUCER, PRESS., GASEOUS, OUTLET, GLYCOL EVAPORATOR
2.45	1	TRANSDUCER, TEMP., OUTLET, SPACE RADIATOR
	1	SENSOR
	1	AMPLIFIER
		GROUP III - COMMAND MODULE PRESSURE & TEMP INSTRUMENTATION
3.40	1	TRANSDUCER, PRESSURE, CABIN TOTAL, COMMAND MODULE
		GROUP IV - OXYGEN SUPPLY INSTRUMENTATION
4.40	1	TRANSDUCER, FLOW, OXYGEN SUPPLY
4.41	1	TRANSDUCER, PRESS., OUTLET, REGULATOR, OXYGEN SUPPLY
4.42	1	TRANSDUCER, TEMP., COMMAND MODULE, AIR, CABIN AVERAGE
	1	SENSOR
	1	AMPLIFIER
		GROUP V - WATER SUPPLY SYSTEM INSTRUMENTATION
5.40	1	TRANSDUCER, PRESS., QUANTITY MEASUREMENT, WASTE WATER TANK
		GROUP X - POWER SUPPLY
10.40	1	POWER SUPPLY, TRANSDUCER, TEMPERATURE (USED WITH ITEMS 1.40, 2.41, 2.42, 2.45, 4.42)

gure 3-10. Service and Command Module Cryogenic Storage, EPS and ECS Integrated Systems

the cryogenics supply to the Environmental Control and Electrical Power system. The general system reliability logic diagrams employed in the system reliability analysis are shown in Figures 3-11 and 3-12.

Mission success reliability has been calculated on a 14-day duration. The abort criteria for such a mission is that the mission must be aborted whenever an additional failure could result in the loss of the spacecraft. To achieve mission success with this criteria, all singularly redundant critical components must be considered in series for mission success until lunar surface stay is completed.

After the spacecraft has proceeded to a transearth phase, the mission objectives will have been achieved if safe return to earth is accomplished. Therefore, after lunar surface stay is completed, the system may be considered in a redundant mode of operation because a single failure no longer aborts the mission.

The predicted probability of mission success and crew safety, using state-of-the-art failure rates, exceeds the apportioned values for a 14-day mission. The time allocation for mission success calculations for the 14-day mission is 147 and 189 hours for the series and redundant-operating modes, respectively.

The time allocation for mission success calculations for the 14-day mission is 147 and 189 hours for the series and redundant-operating modes, respectively. The logic diagram for mission success for the 14-day mission is presented in Figure 3-13. The mathematical models used to compute mission success and crew safety are presented in Figure 3-12. Critical failure modes and their effects on the mission are depicted in Table 3-15, Failure Effects Analysis, located at the end of this section. The predicted state-of-the-art failure rates for these critical failure modes are presented in Table 3-14. The predicted probability of mission success and crew safety, using state-of-the-art failure rates, exceeds the apportioned value for a 14-day mission.

SPECIAL STUDIES

Single Versus Dual Supply Line Concept

An investigation was made to determine if a single supply line could be used instead of the dual supply line now employed in the simplified cryogenic storage system.

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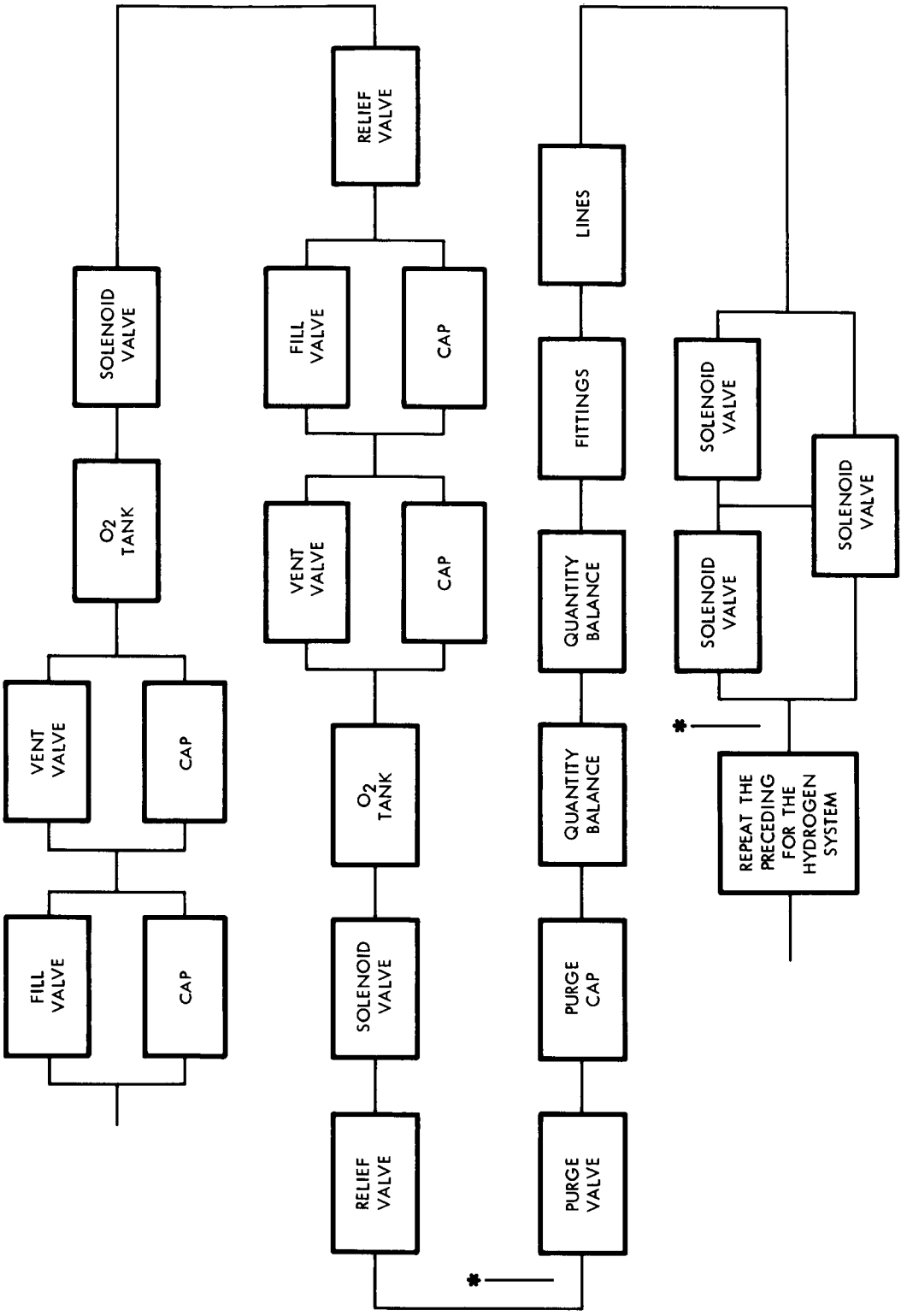

Figure 3-11. Cryogenic Storage System Mission Success Mode Logic Diagram

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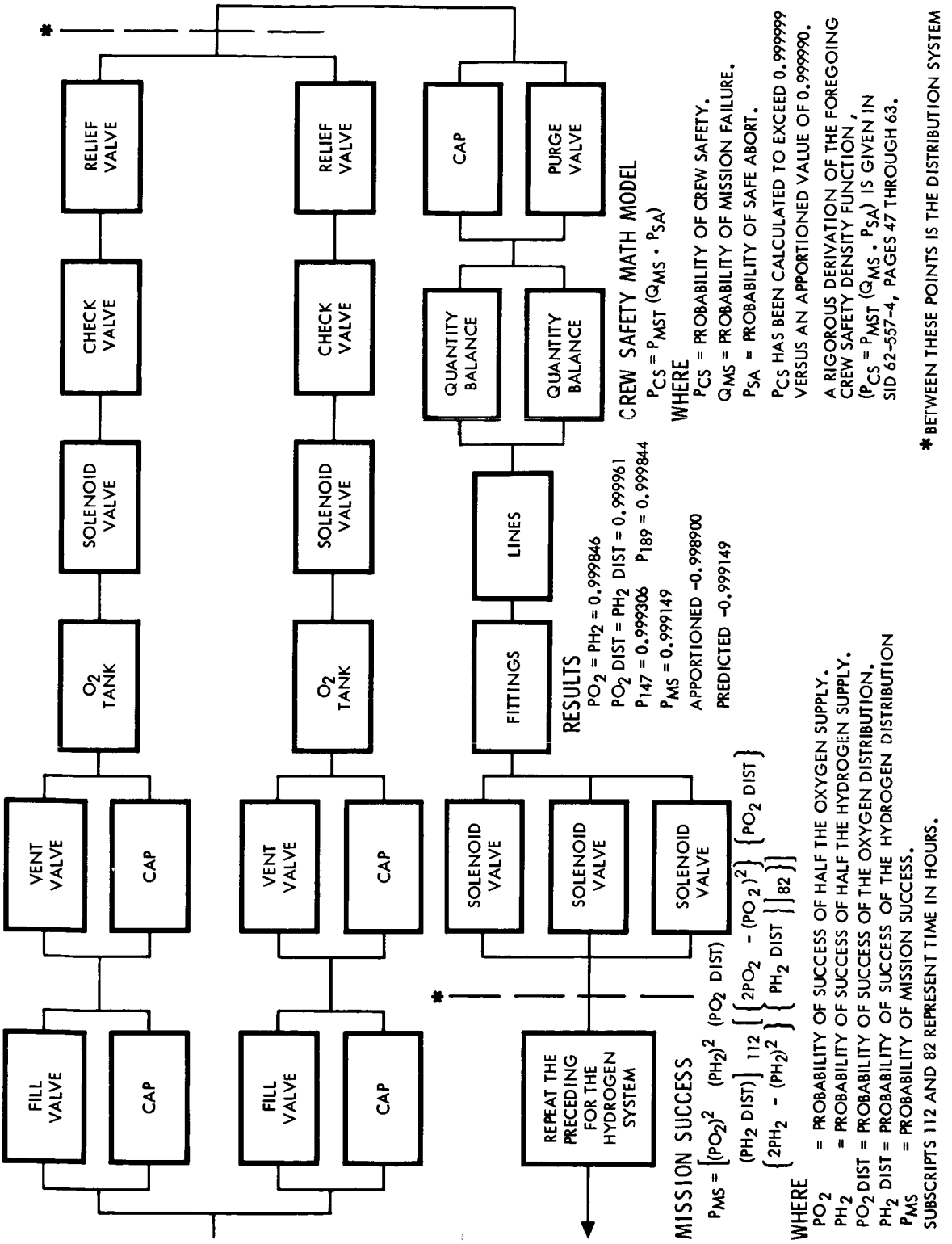
RELIEF VALVE
CHECK VALVE
SOLENOID VALVE
O2 TANK
VENT VALVE
CAP
FILL VALVE
CAP
RELIEF VALVE
CHECK VALVE
SOLENOID VALVE
O2 TANK
VENT VALVE
CAP
FILL VALVE
CAP
CAP
PURGE VALVE
QUANTITY BALANCE
QUANTITY BALANCE
LINES
FITTINGS
SOLENOID VALVE
SOLENOID VALVE
SOLENOID VALVE
REPEAT THE PRECEDING FOR THE HYDROGEN SYSTEM
RESULTS
PO2 = PH2 = 0.999846
PO2 DIST = PH2 DIST = 0.999961
P147 = 0.999306 P189 = 0.999844
PMS = 0.999149
APPORTIONED -0.998900
PREDICTED -0.999149
CREW SAFETY MATH MODEL
PCS = PMST (QMS • PSA)
WHERE
PCS = PROBABILITY OF CREW SAFETY.
QMS = PROBABILITY OF MISSION FAILURE.
PSA = PROBABILITY OF SAFE ABORT.
PCS HAS BEEN CALCULATED TO EXCEED 0.999999 VERSUS AN APPORTIONED VALUE OF 0.999990.
A RIGOROUS DERIVATION OF THE FOREGOING CREW SAFETY DENSITY FUNCTION, (PCS = PMST (QMS • PSA) IS GIVEN IN SID 62-557-4, PAGES 47 THROUGH 63.
* BETWEEN THESE POINTS IS THE DISTRIBUTION SYSTEM
MISSION SUCCESS
PMS = [(PO2)2 (PH2)2 (PO2 DIST) (PH2 DIST)] 112 {{2PO2 - (PO2)2} {PO2 DIST} {2PH2 - (PH2)2} {PH2 DIST}}82
WHERE
PO2 = PROBABILITY OF SUCCESS OF HALF THE OXYGEN SUPPLY.
PH2 = PROBABILITY OF SUCCESS OF HALF THE HYDROGEN SUPPLY.
PO2 DIST = PROBABILITY OF SUCCESS OF THE OXYGEN DISTRIBUTION.
PH2 DIST = PROBABILITY OF SUCCESS OF THE HYDROGEN DISTRIBUTION
PMS = PROBABILITY OF MISSION SUCCESS.
SUBSCRIPTS 112 AND 82 REPRESENT TIME IN HOURS.

Figure 3-12. Cryogenic Storage System Crew Safety Mode Logic Diagram

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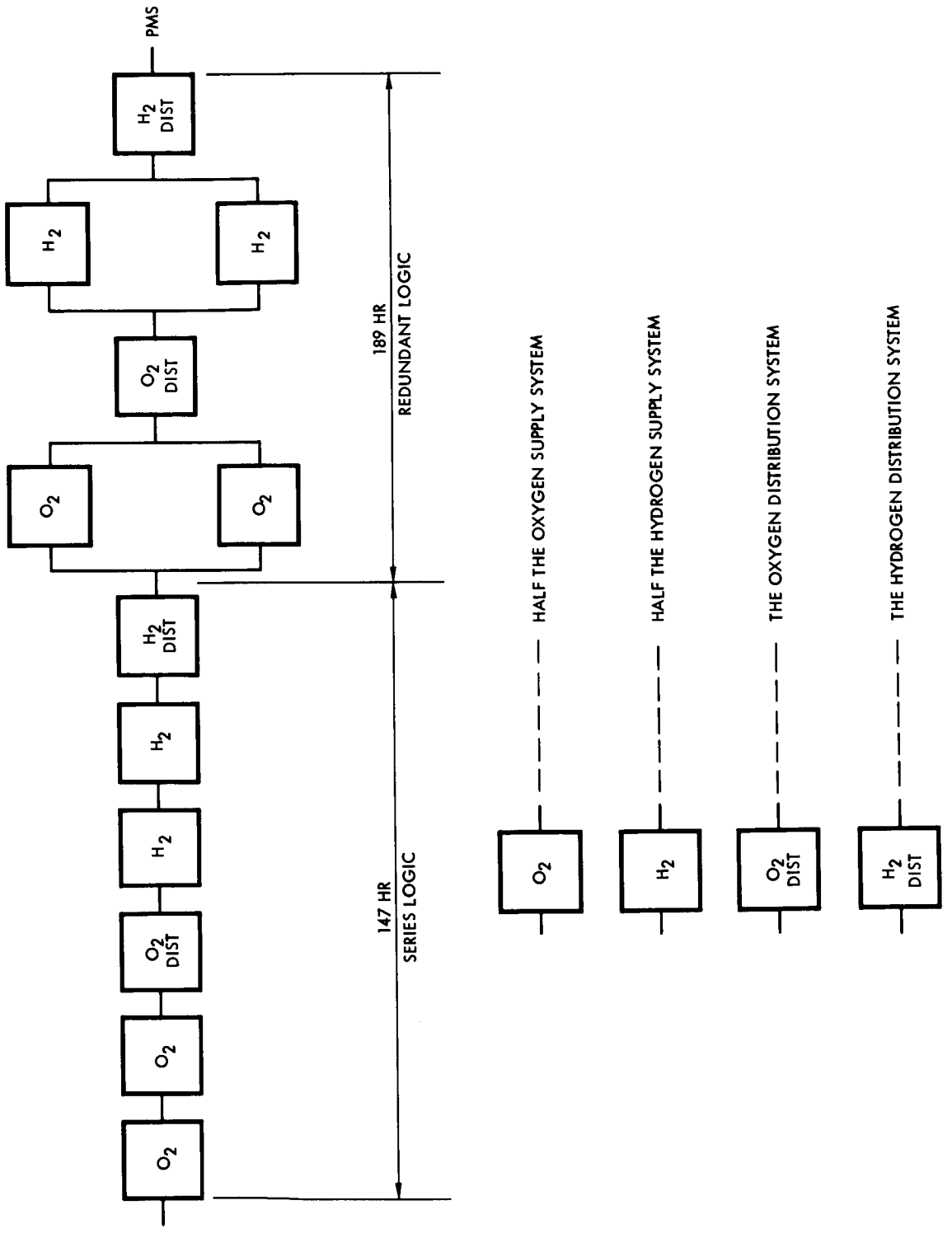

Figure 3-13. Mission Success (14-Day Mission) Logic Diagram

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Table 3-14. Predicted Failure Rates and Reliability Values

Component	Failure Rate Per 10^6 Hr *	Critical Mode Failure Rate Per 10^6	Reliability Values		
			t = 147 (hr)	t = 189 (hr)	t = 336 (hr)
Tank	0.08	0.08	0.999988	0.999985	0.999973
Solenoid valve	4.60	1 Cycle standby	0.999999+	0.999999+	0.999999+
Relief valve	3.90	Fail open 0.96	0.999859	0.999819	0.999677
Fill valve	0.17	Fail open 0.01	0.999999	0.999998	0.999997
Lines	0.0017	0.0017	0.999999+	0.999999+	0.999999+
Fittings	0.00077	0.00077	0.999999+	0.999999+	0.999999+
Quantity balance (S)		0.25	0.999963	0.999953	0.999916
Quantity balance (D)		0.86	0.999999+	0.999999+	0.999999+

S = Series

D = Redundant

*State of the art

Three considerations in favor of the dual line were investigated. One consideration is that it is possible to have the check valves located as far as possible from the source of cold fluid to avoid excessive cycling of the valves, which would result in excessive wear and higher failure probability. A second consideration is that the check valves need not be installed in an opposed configuration in the same line. Installing opposed valves in the same line results in high pressure surges that will cause the valves to alternately slam into the valve seats. The third consideration is the additional measure of safety gained by a completely separate system. If one line should rupture or break, the crew can return on the gas supplied by the second line. As a result of this investigation, it was determined that the dual line offers more advantages from a reliability standpoint than does the single line. It was recommended that the present design remain as a part of the over-all cryogenic storage system configuration.

Single-Tank Cryogenic Storage System for Airframe 009

A reliability analysis was performed to determine the effect on crew safety if a single-tank cryogenic storage system is used in Airframe 009. The fuel cells will be carrying a dummy load throughout the mission and electrical power will be supplied by battery. Based on these considerations, a single-tank hydrogen storage system will be adequate, since it will not affect crew safety.

The oxygen storage system was investigated further because crew safety is directly dependent on oxygen supply. The system studied contained enough components to assure check-out and reactant delivery. The system contained the same redundant control circuitry as the dual-tank system. The results of the analysis showed the system reliability to be 0.999998 for the twenty-six minute mission time. This high reliability is a result of high reliability components, minimum number of components, and short mission time.

Reliability Engineering, considering the preceding analysis, approved the use of a single-tank cryogenic storage system in Airframe 009. The oxygen system schematic is shown in Figure 3-14.

In the event that at some time in the future a requirement is generated to use fuel cells for purposes of supplying primary electrical power, a reliability analysis of dual hydrogen supply system installation and operation must also be generated.

Deletion of Single-Tank Five-Minute Emergency Requirement

A reliability study was performed to determine the probability of a command module puncture and an oxygen storage system failure occurring within the mission time period. This situation would require the operation of a single-tank system under emergency flow conditions for 5 minutes. The analysis showed the probability of occurrence of the puncture and the system

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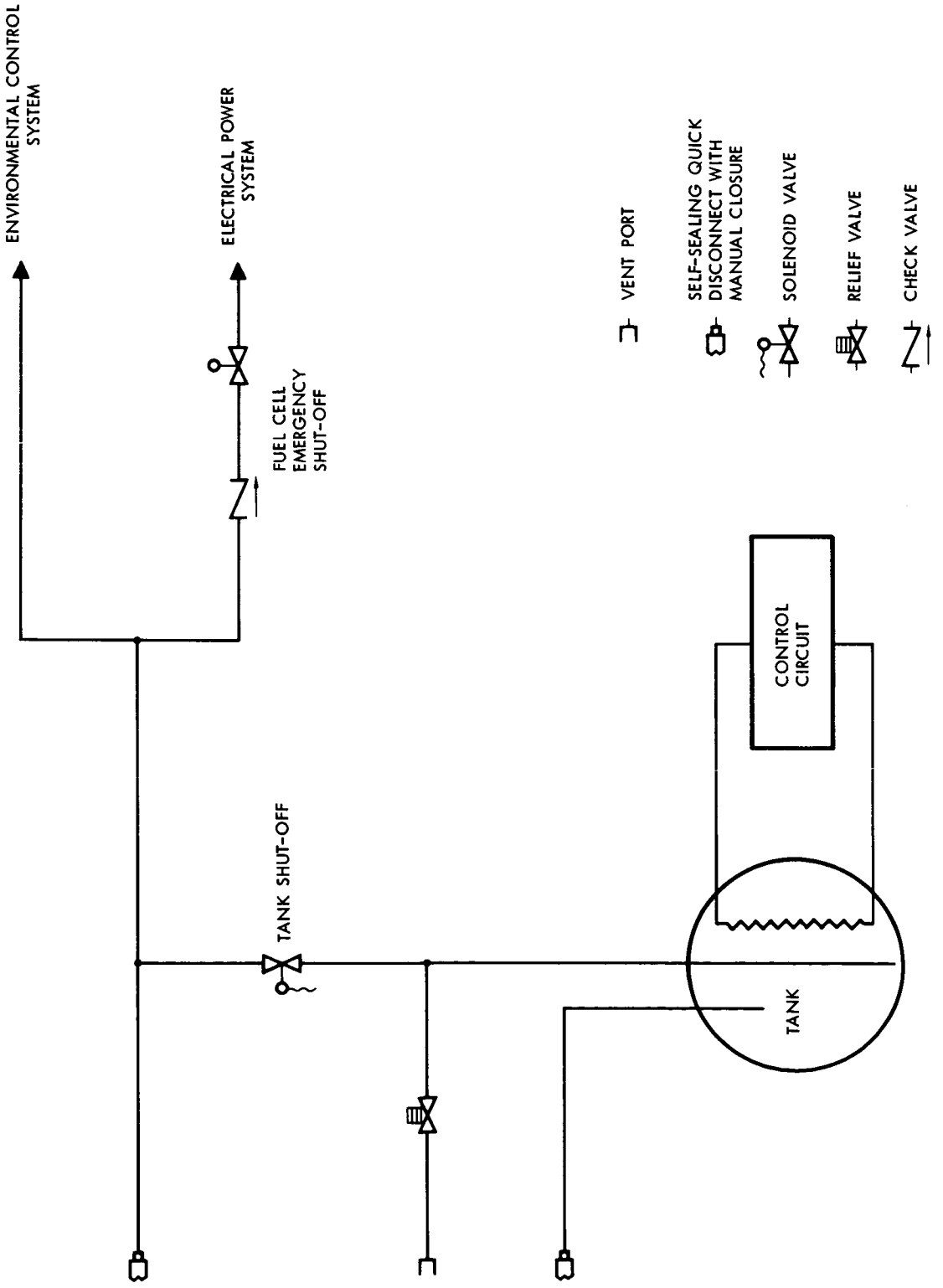

Figure 3-14. Airframe 009 Oxygen System Schematic

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failure within the mission time period of 336 hours to be less than one occurrence in a million missions.

In view of the high improbability of the occurrence, Reliability Engineering concurred with the deletion of the requirement for a capability to maintain command module pressure for 5 minutes employing only one O_2 tank.

Command Module Repressurization

Beech Aircraft Corporation has suggested resizing the heaters to provide the one-hour repressurization utilizing both cryogenic oxygen storage tanks. The heaters that are currently utilized to maintain pressure in the cryogenic storage system provide sufficient heat for repressurization of the command module in one hour, employing one cryogenic oxygen storage tank. Beech indicated that the required electrical power will be half the present value if both tanks are utilized for repressurization.

Based on the following considerations, Reliability Engineering recommended that the heaters be resized to allow repressurization by both cryogenic oxygen tanks in one hour.

1. Repressurization of the command module is required only after a command module puncture or an extra-vehicular operation.
2. Extra-vehicular operation is not part of the mission plan. Any failures occurring will have to be handled by abort procedure because there is no provision for service module or command module external repairs.
3. The probability of a command module puncture coupled with an oxygen subsystem failure is less than one occurrence in a million missions.
4. In the event of a depressurization of the CM, the crew will don space suits. With crew in suits the length of time taken to repressurize the CM is not critical.
5. Reliability of the heater system will be enhanced if power requirements are lowered; for example, lower power requirements would result in extended heating intervals and fewer switching operations.

TEST PROGRAM

Qualification Test

Information was submitted for Volume III, Ground Qualification Tests, of the General Test Plan Research and Development for Apollo Spacecraft (SID 62-109). The data included a definition of the system test program, a system logic diagram, a presentation of system criticality versus hardware requirements, and a qualification test schedule with test specimen utilization.

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Problem Areas

During the development of the 17-inch pressure vessels, a problem developed relative to the interpretation of the X-ray technique and check of proper alignment of the weld beam. In addition, a failure resulted during the proof pressure testing of an inconel pressure vessel. Inspection revealed microscopic cracks in the weld. Similar cracks were discovered in the other vessels tested, including four vessels fabricated from Inconel 718. The following action is being taken by the supplier.

> To provide for adequate inspection of the peripheral weld of the inner pressure vessel, the joint was redesigned from a modified butt joint to a true butt joint. This was accomplished by the removal of the weld back-up section originally designed into the joint. The back-up section was designed into the joint to trap the weld's splatter. Since it had been determined that the weld's splatter could be tolerated within the tank by the use of a filter on the feed line, the back-up section was removed to prevent interference by this section during the X-ray of the weld joint. To further facilitate X-raying of the joint, the middle section of the heaters in the plane of the joint was removed by redesign of the heaters. The peripheral joint is now clear of any obstruction for X-ray purposes. The inner side of the peripheral joints is visible for visual inspection. Visual gauging of the weld penetration has been designed into the joint by a machine land on the inner surface. Visual inspection of the inner side of the peripheral weld is accomplished by the use of a boroscope. The inspection procedures now available by redesign of the joint and the tank heaters are considered to be adequate.

During receiving inspection checkout, one of the thermoswitches was erratic. An investigation conducted by Engineering and Reliability determined that the vendor-supplied thermoswitch was unreliable when used at liquid hydrogen temperatures. Two approaches to this problem are being made. One is to eliminate the switch from the design of the heater, and the other requires a vendor investigation to secure switches suitable for use at liquid hydrogen temperatures. To date, Texas Instruments is the only vendor contacted as a possible supplier of these switches. A shipment of five switches for test purposes is on the way to Beech. These will be checked at liquid hydrogen temperatures. Further follow-up will be made.

Hamilton Standard is having trouble meeting the leakage requirements of the procurement specification. The valve design state-of-the-art at Hamilton Standard will be checked for clues to leakage problems. Beech Reliability and Engineering are reviewing the valve leakage requirements in terms of the simplified system.

The density sensor accuracy is 0.5 percent full scale from 25 percent to zero percent tank volume and 1 percent full scale accuracy within the NAA procurement specification requirements for tanks that are not stratified. Beech has determined that stratification of 10 degrees would cause gauge reading errors that are beyond the allowable limits of NAA's procurement specification. Beech is investigating equilibration, other means of quantity measurement, and system management.

Development Tests

1. Tube Connector Vibration Test. The test setup was changed to allow for injection of helium gas into the liquid hydrogen line used to cold soak the specimen. This allowed leak detection to be monitored by the mass spectrometer method. The connections tested were MC, Parker Lo-torque, and brazed. Test results are in preparation.

2. O_2 Heater Vibration Test. Component acceptance testing was completed on the 10 heater hemispheres. Vibration testing began during March 1963.

3. Burst Test of Rupture Discs. Commercial discs were obtained with testing proceeding on schedule. Three Inconel X discs have been tested. Titanium discs have not yet been received from the manufacturer.

SUBCONTRACTOR MANAGEMENT

Subcontractor Coordination

Following an in-house design review by Beech on both the hydrogen and oxygen vessels, a reliability design review was held on 13 January 1963 by Beech and NAA/S&ID. This review occurred after the completion of the 17-inch pressure vessel test program. The significant results of the tank design review are that the weld joint design on the tank will be redesigned, and the inspection procedures for inspecting the joint will be changed.

Subcontractor Activities

Design Review

A design review between Beech and Simmonds Precision Products on the density sensor was held 14-15 January 1963. The significant results of this review are as follows: Beech gave Simmonds an explanation of reliability required, in which the elimination of connectors was considered in

order to improve sensor reliability. Simmonds made suggestions to Beech for the reduction of weight in their design, which Beech will consider. Simmonds asked for answers to four questions.

What is to be done with the output signal?

What is the input impedance of the readout device to be used in the command module?

Can the signal conditioner use single ground in combination with power ground?

How much ripple is permissible in the output signal?

Answers to these questions have been transmitted by Beech to Simmonds. Simmonds furnished Beech calculations on the effect of temperature and stratification, which will be studied and confirmed by Beech as to their correctness. Beech confirmed that zero volts output indicates zero usable fluid remaining. Simmonds' design layouts of the density sensors were reviewed and approved by Beech.

Design and Development

In addition to the release of the new simplified system schematic, calculations for insulation thickness procedure, heat leak through coil cover, and ullage requirements were documented. The technical evaluation of the temperature probe signal conditioner and the preliminary thermodynamics analysis for the Apollo tank system were completed.

A report has been prepared concerning the response, characteristics, and results of probes and thermocouples, with additional recommendations. Work is continuing on the flat plate insulation samples to determine outgassing characteristics and K[1] values on the flat plate calorimeter. The nine-inch diameter insulation vibration model test was completed, and the test report has been rough drafted. An electron beam development program was prepared, including a drawing for welding samples with misaligned beam weld for test coupons.

A preliminary stress report was written, interpreting the test reports of the four 17-inch diameter pressure vessels. The dynamic buckling frequency for external Inconel X shell was calculated. The evaluation of the vibration fatigue test on the nine-inch diameter insulation model was completed.

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[1] K = Thermal conductivity of the material

The reliability engineering evaluation of the over-all test program is continuing. Work is progressing in the statistical evaluation of the test program and establishment of the mathematical model to be used in determining reliability.

General

Development test reports of the tests that are in progress are being reviewed for release. The development test report on the 17-inch pressure vessels is in the system at this time and has not been released. Beech Reliability Plan, BR 13707, Part I, is currently being revised so that it reflects the latest changes to the procurement specification. Part II of the report is being incorporated into BR 13705, Test Plan, per North American MC 999-0025 instructions.

PLANNED ACTIVITIES

1. Considerable effort will be expended in the area of subcontractor coordination and documentation. Increased coordination and reliability direction is expected to lead to early approval of subcontractor type I documents concerning reliability.

2. Analyses will be performed to further simplify and improve the reliability of the electrical control circuitry.

3. Reappraisal of the system due to changes that may be necessary for LEM integration is expected to commence during the next reporting period.

4. The fill and vent coil testing will commence when test items are received. Vibration fixture is scheduled for delivery during the next reporting period.

Table 3-15. Cryogenic Storage System Failure Effects Analysis

Item	Drawing Number	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	System Performance	Apollo	
Fill valve Vent valve Purge valve			Allows passage of fluid during the countdown. Must not leak throughout mission.	Leakage	Launch to SM disconnect	Erosion of seal	Loss of gas in excess of allowable losses	Abort mission	Close good supply. Use as much as possible of the failed system. Complete abort on good system.
Tank			Stores fluid	Leakage	Launch to SM disconnect	Wall rupture	Loss of gas in excess of allowable losses	Abort mission	Close good supply. Use as much as possible of the failed system. Complete abort on good system.
						Burst			Close failed system. Use good system to complete abort.
Relief valve			Relieves excess system pressure	Leakage - fail to re-seat	Launch to SM disconnect	Erosion of seal	Loss of gas in excess of allowable losses	Abort mission	Close good supply. Use as much as possible of the failed system. Complete abort on good system.
				Fail to open when required	Launch to SM disconnect		Excess pressure in system	None	None. The relief valve is backed up by relief valves in the ECS and EPS.
Solenoid valve (system shut-off)			System checkout prior to launch—locked open during mission	External leakage, case cracking, or rupture	Launch to SM disconnect	None	Loss of gas in excess of allowable losses	Abort mission	Close good supply. Use as much as possible of the failed system. Complete abort on good system.
Quantity and pressure control	Beech 13532-0030	N/A	Maintains system pressure and ensures the use of equal quantities of fluid from both tanks by controlling the amount of heat applied to the fluid in the tanks.	Open circuit (wiring)	Launch to SM separation	None	Pressure loss in one of the tanks and subsequent depletion of the other	Loss of redundant supply capability	Activate manual override circuit and maintain surveillance.

Table 3-15. Cryogenic Storage System Failure Effects Analysis (Cont)

Item	Drawing Number	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	System Performance	Apollo	
				Pressure switch fails open	Launch to SM separation	Component rendered useless	Pressure loss in one of the tanks and subsequent depletion of the other	Loss of redundant supply capability	Activate manual override circuit and maintain surveillance
				Pressure switch fails closed	Launch to SM separation	Component rendered useless	One tank will overpressure and be depleted before the other.	Loss of redundant supply capability	Activate manual override circuit and maintain surveillance
				Resistor, open failure	Launch to SM separation	Component rendered useless	Pressure loss in one tank and subsequent depletion of the other	Loss of redundant supply capability	Activate manual override circuit and maintain surveillance
Quantity and pressure control				Resistor, short failure	Launch to SM separation	Component rendered useless	One tank will overpressure and be depleted before the other.	Loss of redundant supply capability	Activate manual override circuit and maintain surveillance
				Resistor, out of tolerance	Launch to SM separation	Component's usefulness marginal	This is unpredictable. The effects may range from sluggish operation of circuit to complete open condition.	Variation of tank quantities out of 15 pound tolerance	Situation is marginal. Decision will have to be made on the basis of the seriousness of the problem at the time.

Table 3-15. Cryogenic Storage System Failure Effects Analysis (Cont)

Item	Drawing Number	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	System Performance	Apollo	
				Heater, open failure	Launch to SM separation	Component rendered useless	Pressure loss in one tank and subsequent depletion of the other	Loss of redundant supply capability	Activate manual override circuit and maintain surveillance under abort conditions.
				General SCR failure	Launch to SM separation	Component's usefulness marginal	This is unpredictable. The effects may range from sluggish operation of circuit to complete open condition.	Variation of tank quantities out of 15 pound tolerance	Situation is marginal. Decision will have to be made on the basis of the seriousness of the problem at the time.
				Warning light burn-out	Launch to SM separation	Component rendered useless	None	None	Will have to monitor pressure, temperature and quantity gages.
Quantity and pressure control				General transistor failure	Launch to SM separation	Component's usefulness marginal	This is unpredictable. The effects may range from sluggish operation of the circuit to complete open condition.	Variation of tank quantities out of 15 pound tolerance	Situation is marginal. Decision will have to be made on the basis of seriousness of the problem at the time.
				Short circuit (wiring)	Launch to SM separation	Component rendered useless	Loss of heater control or over-pressure and subsequent depletion of one tank before the other.	Loss of redundant supply capability	Activate manual override circuit and maintain surveillance

Table 3-15. Cryogenic Storage System Failure Effects Analysis (Cont)

Item	Drawing Number	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	System Performance	Apollo	
Lines and fittings				Leakage	Launch to SM separation	Structurally weakened	Loss of gas	Insufficient gas to compute the 336 hour mission	Detect line or fitting and isolate it from the rest of the system by valve actuation if possible. Abort mission and maintain surveillance.

COMMAND MODULE REACTION CONTROL

SUMMARY

Apportioned reliability for the command module reaction control subsystem (CM RCS) is no longer being met with the present design. With present procurement specification components, a predicted probability of 440 failures per 10^6 missions would have to be accepted as compared with the required apportioned value of 40 failures per 10^6 missions. This inability to meet the apportioned system reliability is a result of an added operating requirement. As long as an adequate impact attenuation system was included in the spacecraft design, the only operating requirement of the CM RCS was attitude control. Because of the deletion of the impact attenuation provisions, it will be necessary to expend all CM RCS reactants prior to earth impact. For this programmed burning (i.e., consuming reserve propellants) method of operation, further design changes do not significantly improve the reliability of the system.

The latest CM RCS design configuration is shown in Figure 3-15. This configuration differs from earlier configurations in that dump, bypass, and interconnect provisions (3, 5, and 6 through 8, Figure 3-15) have been added. The remaining components were identified in the logic diagram in SID 62-557-4. Because of these design and functional changes, the system operational analysis, logic analysis, failure effects analyses, and reliability predictions must all be updated. These analyses have been initiated and will incorporate the latest design refinements.

ANALYSIS

Failure Mode Analysis

An apportionment of component mission failures among the applicable modes of failure for each of fourteen types of CM RCS components was completed (Table 3-16). This apportionment was necessary to permit an accurate, detailed analysis of the ability of the revised design of the CM RCS to consume its hypergolic reactants prior to earth impact.

Failure Indication Criteria

At present, the CM RCS has no mission success (as differentiated from crew safety) reliability requirement or apportionment. This is the result of an early analysis which indicated that the crew would have no indication of a failure before the system was required at earth entry—too late for an abort.

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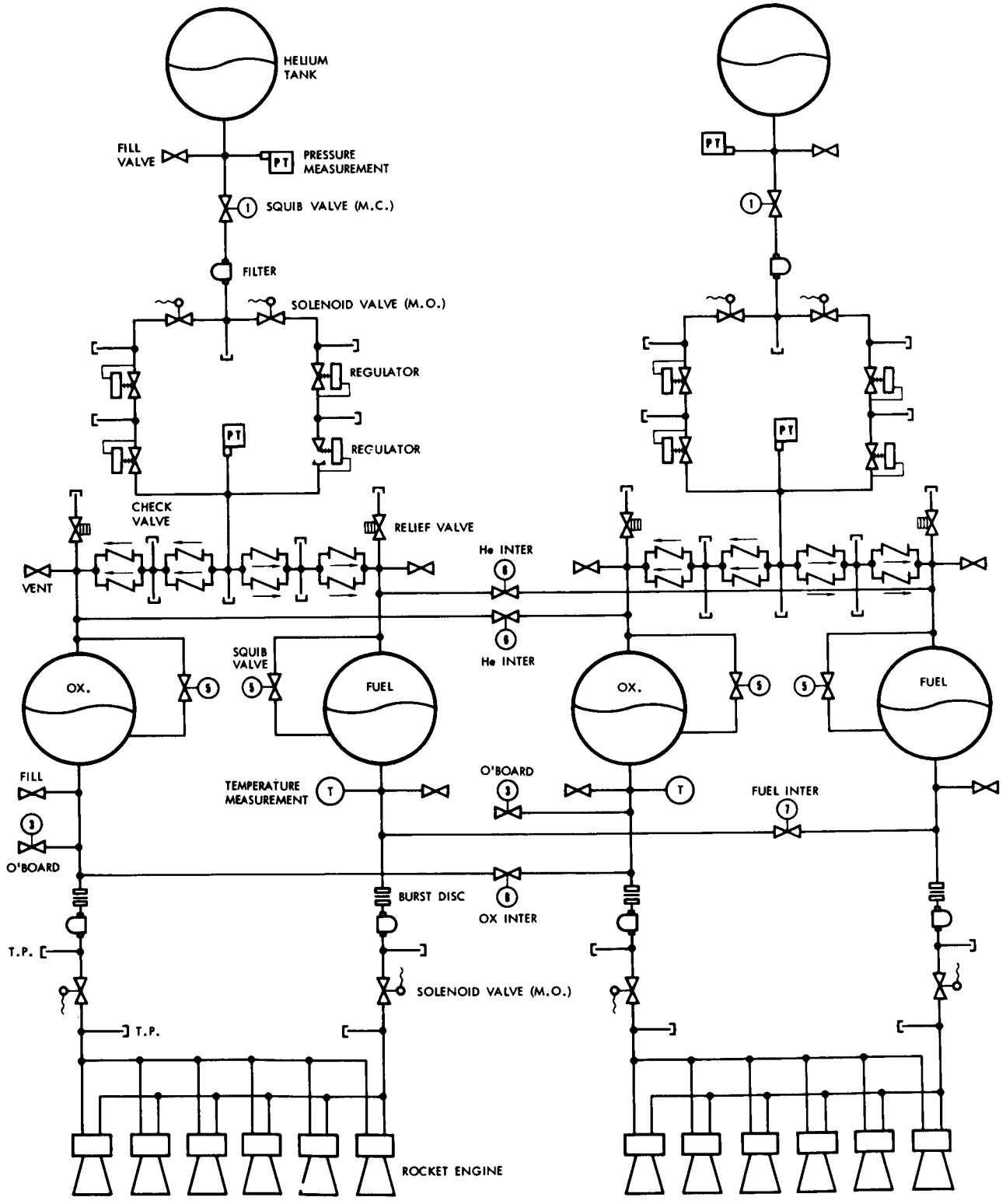

Figure 3-15. Revised Command Module Reaction Control System

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Table 3-16. Failure Analysis for Programed Burning

Reactant Supply Components	Failures For 10^6 Missions (Procurement Requirements For Each Component)		Component Failure Modes	Series Failures (Circled Modes)	
	NORMAL	ABORT		Normal Mission	Pad Abort
Helium check valve (8)	90	85	1. Fail closed 2. Fail open (3.) External leak	5	5
Helium vent valve with cap (4)	1	1	(1.) External leak or rupture	1	1
Helium burst disc (4)	17	8	(1.) Burst high 2. Burst low 3. Internal leak (4.) External leak	10	5
Positive expulsion tank (4)	100	1	(1.) Rupture (2.) External leak (3.) Support failure 4. Internal leak 5. Expulsion failure	10	1
Reactant fill valve with cap (4)	1	1	(1.) Rupture (2.) Leak	1	1
Reactant burst disc (4)	17	8	(1.) Burst high 2. Burst low (3.) External leak	10	5
Reactant filter (4)	12	11	1. Fail to trap (2.) Clogged (3.) External leak	4	4
Reactant fittings (60)	34 (Total)	34 (Total)	(1.) External leak or rupture	34 (Total)	34 (Total)
Reactant tubing (33)	40 (Total)	40 (Total)	(1.) Rupture (2.) Pinch or bend	40 (Total)	40 (Total)
Reactant solenoid valves (4)	239	230	1. Fail to operate 2. Premature operation (3.) External leak 4. Internal leak	15	14
Rocket engines (12)	520		(1.) Explosion 2. Fire 3. Excessive ablation 4. No start 5. Improper ratio (6.) External leak	70	
Helium squib interconnect valves (2)	520	230	1. Fail to operate (2.) External leak 3. Internal leak 4. Premature operation	5	2
Reactant squib interconnect valves (2)	520	230	1. Fail to operate (2.) External leak 3. Internal leak 4. Premature operation	5	2
Depressurization and dump squib valves (2)	520	230	1. Fail to operate (2.) External leak 3. Internal leak 4. Premature operation	5	2

A preliminary analysis of CM RCS failure indication criteria was undertaken to determine the feasibility of adding a mission success requirement for the system.

Redundant tank pressure monitoring, leak detection provisions, and sensitive quantity measuring devices are under investigation.

SPECIAL STUDIES

A comprehensive CM RCS reliability study of a critical problem area was completed. This study indicated that, as a consequence of present reliability, design, and operating requirements, a condition exists which would jeopardize the survival of the Apollo crew upon landing. These requirements are as follows:

1. A 100-percent reserve of reaction control propellants (reliability redundancy)

2. A 360-degree peripheral distribution of hypergolic propellants and the removal of impact attenuation (design requirements)

3. An earth impact with the crew in the spacecraft and remaining there during a subsequent cooling period (operating criteria)

The resulting danger to the crew is the probable occurrence at impact of a catastrophic fire or an explosion and/or the release of extremely hazardous, toxic fumes.

Three possible solutions have been investigated; they are isolation of the reactants, separation of the crew from the reactants, and neutralization of the reactants. This analysis, however, has not indicated a satisfactory solution to the problem.

Without the propellant reserve, the crew safety reliability would decrease from 0.999968 (32 failure per 10^6 missions) to approximately 0.99 (10,000 failures per 10^6 missions). In addition, the tanks containing the reactants would contribute to the determination of the point of impact because of their weight, the impact force could be in excess of 75 g (estimate based on boilerplate drop tests), and the heat shield would act as a heat source at impact with its surface temperatures estimated in excess of 500 F at the exterior surface and 150 F at the interior surface. A complicating factor is that the reactant alone in either liquid or vapor form is hazardous. If a fire should occur at any time during or after impact, the delay caused by the requirement that the three crew members encumbered by spacesuits must disembark through the 30-inch overhead tunnel-hatch would be exceedingly dangerous.

Subsequent supporting analyses were completed that resulted in twelve recommendations of possible solutions to the problem. At a design review board meeting, the problem was presented, and the board recommended evaluation of programmed burning. The programmed burning operating change with its subsequent design changes was carefully analyzed, and a series of detailed reliability studies was completed. Improved component reliability data and failure mode analysis and apportionments were used in these studies. The studies evaluated the ability of the complete system, with design revisions, to perform four separate but related functions. Those functions are attitude control of the spacecraft, consumption of reserve reactants during normal reentry through programmed burning, independent jettisoning of reserve reactants during pad abort, and purging the extensive complex of tubing and components prior to impact. The study results are contained in Table 3-17.

Table 3-17. Reliability Evaluation of Proposed RCS Design Revisions

Item	Crew Safety Reliability	Percent of Missions Attempted
1. Reliability of the system before design changes in performing attitude control only	0.999968	
2. Reliability of the system as revised in performing attitude control only	0.999974	
3. Reliability of the system as revised to perform attitude control, burning of reserve reactants, and purging	0.9996	95
4. Reliability of the system as revised to perform only jettisoning of reserve reactants during pad abort including purging	0.9989	5
5. Resultant reliability for the combined system (i.e., average mission reliability for the revised system considering all four functions)	0.99956	100
6. Required system reliability	0.999960	100

The system reliability requirement shown in Table 3-17 is less than 40 failures per 10^6 missions, but if the absence of reactants at impact is to be a crew safety requirement, the predicted system reliability is 440 failures per 10^6 missions. This is true even if the requirement demanded by the procurement specifications is achieved. Thus, an additional elevenfold improvement factor requirement in excess of procurement specification development would be needed.

A system design analysis was completed that investigated the possible effectiveness of additional changes in the design of the CM RCS. These additional changes would be made in an effort to increase the system reliability during performance of its four required functions. The results of this analysis indicate that neither increasing nor decreasing the number of systems or components would significantly improve the composite reliability. This apparent inconsistency is because one function requires reactants at reentry and another requires the absence of reactants at earth impact.

PROBLEM AREAS

The current ability of the CM RCS, with the recent revisions, to perform its four required functions is reflected in a reliability of 0.99956 (440 failures per million missions). The system apportionment, however, is no more than 40 failures per 10^6 missions out of a total Apollo spacecraft apportionment (excluding LEM) of 420 failures per 10^6 missions.

The logical parameters to consider in an effort to remove this deficiency are to improve the programmed burning system design, increase component product improvement, revise the ground rules, or supplement programmed burning as a partial solution to the impact safety problem. System redesign would be ineffective, as explained earlier. Additional component development would be impractical because the present CM RCS procurement specification component reliability values require product improvement factors between five and fifty times the current state-of-the art. These same procurement specification reliability values provide the system reliability of 0.99956 noted above rather than the system reliability requirement of 0.99996.

The ground rule parameter would entail either reduction of the crew safety reliability requirement or spacecraft reapportionment. Safety requirement reduction would permit the loss of more than one crew per thousand while spacecraft reapportionment would mean that the CM RCS would require 440 failures per 10^6 missions, while only 1000 failures per 10^6 missions are permitted for the entire rocket-spacecraft-LEM combination. Neither choice is desirable.

Supplementing programmed burning as a partial solution to the problem is the only remaining parameter. This could mean separation of the crew from reactants either by crew bail-out after main parachute deployment, which would require a bail-out door and individual parachutes, or by encapsulated ejection of the reactants after main parachute deployment, which would require fairly extensive structural redesign. Less effective steps than these could include some combination of the following:

1. Replacing impact attenuation and increasing structural integrity of the system components and mounts
2. Landing in water
3. Encapsulating foam
4. Gelling the reactants
5. Providing a blow-out door and NO_2 indicator
6. Shock mounting tanks and tubing

Results of continued investigation to resolve these problems will be the subject of subsequent reports.

TEST PROGRAM

Qualification Test

Data was submitted for inclusion in Volume III (Ground Qualification Tests) of SID 62-109. The data included a definition of the systems test program, a system logic diagram, a presentation of system criticality versus hardware requirements, and a qualification test schedule with test specimen utilization.

General Test Program Status

The Rocketdyne development test program is approximately 15 percent complete. During one engine test, nozzle burn-through was experienced. Investigation of this problem is currently underway. Engine qualification tests are scheduled to commence in August 1963. Component qualification tests are scheduled to begin in May or June 1963.

Specific Development Tests

The Rocketdyne development test program was initiated on both the propellant valves and the engine. The propellant valves have undergone vibration, life, and temperature exposure. Engine tests were conducted to

determine injector delta pressure, chamber pressure, ignition characteristics, and mixture ratio. Development tests on the engine will continue for various firing durations at altitude, temperature extremes, off-nominal mixture ratio, and low supply pressure. Specific component development tests have not been initiated, but all such tests are to commence next quarter.

Flexible Tubing Tests

Because of certain potentially hazardous characteristics of the reactants in the CM RCS, purging of numerous sections of flexible tubing is required. For this reason, Apollo Reliability Engineering prefers a smooth-bore design for the flexible tubing rather than a bellows, coiled, or similarly interrupted inner surface design. Encouraging results of preliminary tests of a typical design configuration for Teflon tubing have been reported by Engineering Development Laboratory. Because of the possibility of premature pressurization of the reaction control system and subsequent permeation of the Teflon liner of this flexible tubing configuration, it is requested that further testing be undertaken. The specific information requested is the concentration in parts per million (ppm) of NO_2 (per foot of flexible tubing and per cubic foot of contaminated volume) caused by the permeation of N_2O_4 under 360 psia for an operating period of 336 hours. This period must be followed by a 72-hour period at normal vapor pressure.

SUBCONTRACTOR MANAGEMENT

Rocketdyne, manufacturer of the Apollo CM RCS engine, has been informed of the reliability documentation requirements, and has agreed to furnish test data and reports as requested.

A preliminary qualification test plan has been completed, and the plan is now being corrected to incorporate differences required by the design changes (e.g., 10:1 expansion ratio instead of 40:1 and a two-coil solenoid valve instead of a three-coil solenoid).

Compatability tests of the valve materials have shown that the tungsten carbide ball in the valve armature is subject to etching from N_2O_4. A program has been initiated to determine a more suitable material. Early tests indicate that sapphire may be a desirable material.

Several test firings have been made to determine the delta P across the injector at rated chamber pressures, to determine the effects of various mixture ratios, and to measure the char rate. Tests are also scheduled for material compatability and electrical characteristics of the valves.

Work is in progress on the development, reliability assurance, and prequalification test plans.

PLANNED ACTIVITIES

Investigation of the effects on reliability of further CM RCS component product improvement is in progress.

Studies will be initiated to estimate more accurately the probabilities of the spacecraft landing with sufficient impact to exceed the structural integrity limits of the storage and crew compartments, command module reaction control system, or environmental control system; not landing on water; landing so as to damage the only exit; landing resulting in injury to the crew; and landing resulting in fire from sources other than the CM RCS reactants. The studies will also be used to estimate the effect of these factors in the determination of the extent of potential hazard.

SERVICE MODULE REACTION CONTROL

SUMMARY

The service module reaction control subsystem (SM RCS) is the quad system. It consists of four independent quadrant modules, each with an integral propellant supply and four engines. Mission success and crew safety apportionments for the quad system are attained for the criterion that two of the four quadrants must operate successfully.

Further analyses of several configurations have been made to improve the system reliability, to evaluate more stringent system mission parameters, and to investigate beneficial reliability effects on other systems. Modification B of the quad system can provide more flexibility in the SM RCS and improve the reliability of the service propulsion system (SPS). This modification, which consists of the addition of four engines mounted at the rear of the service module, could provide a backup for the SPS during all mid-course corrections and a partial backup for the gimbal actuators during the entire mission.

The reliability values for the quad system and modification B, based upon the present mission parameters, are presented in Table 3-18. These values show the desirability of the modification.

ANALYSIS

A study of the service module reaction control system (SM RCS) was completed by Apollo Reliability Engineering. Several different facets, all having pronounced effect upon the reliability of the SM RCS, were encompassed. The reliability analyses covered the evaluation of various system configurations, the determination of the advantages or disadvantages associated with proposed system modifications, and the investigation of varying mission parameters. The system configurations, modifications, and mission parameters involved in these analyses are described in this section. The analyses indicated that only a limited number of configurations investigated would meet the reliability requirements if the more stringent mission parameters were used in the evaluations. Therefore, as the severity of the mission parameters was decreased, the number of acceptable configurations increased.

Table 3-18. Service Module RCS Reliability Summary

Configurations	Mission Success	Crew Safety	Remarks
Quad	0.998434	0.999989	Assumptions: 1. Translation in all axes is required through LEM transposition docking. 2. Three out of four modules are required until trans-earth inject. 3. Any two out of four modules are required after transearth inject. 4. CM-SM separation can be accomplished with one module.
Modification B	0.999027	0.999998	Assumptions: 1. Translation in all axes is required through LEM transposition docking. 2. Two out of four modules plus all aft engines are required from transposition docking until trans-earth inject. 3. One out of four modules plus all aft engines are required after transearth inject. 4. CM-SM separation can be accomplished with one module.
Apportioned Values	0.996000	0.999970	Note: Although the assumptions for the systems appear to be different, the end results are the same since both provide the same degree of control.

To simplify the system logic equation, the components were independently grouped. The major groups were the helium supply, the fuel supply, the oxidizer supply, and the engine set groups.

The reliability analyses of the various system configurations were accomplished with binomial expansion. All possible combinations of system success and system failure were determined. The various combinations of success terms and failure terms in the binomial equation were associated with either mission success or mission failure. The terms representing mission failure were deleted from the equation, and the remaining success terms yielded the probability of success for the configuration.

Mission success is defined as the probability of successfully completing the mission without a failure that would require abort. Although this definition appears simple, its application can be difficult. The difficulty arises because the system requirements change for different phases of the mission. For example, there is a certain point in the mission where the system requirements necessary to satisfy the mission success criteria are the same as those required for the crew safety criteria. This condition exists when it is no longer possible to modify the mission to return to earth by an alternate route. The time in the mission that this occurs, however, has been subject to interpretation. Depending on the mission parameters, this point could be considered almost anywhere in lunar orbit. As a result of this study, this point is considered to occur at transearth injection.

Crew safety is defined as the probability of successfully returning to earth with no failures that would result in the crew exceeding emergency limits. There is a problem associated with crew safety similar to the difficulty encountered in the application of the mission success definition. There is a lack of definitive information on what constitutes emergency limits and what system requirements would be needed to prevent the crew from exceeding these limitations. At the present time, any two of the four modules are required to prevent the crew from exceeding emergency limits.

The mission profile used in the analyses has a total time of 14 days (336 hours). Since the SM RCS is used for intermittent bursts throughout the mission, the number of mission phases is reduced, and the analyses simplified. The specific portions of the mission that were consolidated are listed in Tables IV through VIII.

These criteria indicate that the present quad system would meet the reliability and weight requirements; however, some of the ground rules have not been validated by the cognizant engineering department and are presently being investigated. The results of this investigation will determine whether or not the quad system is desirable. For example, indications are that additional propellant would be required if three of the four units were used during the translunar portion of the mission.

The quad system, presented in Figures 3-16, 3-17, and 3-18, consists of four completely independent propulsion units located 90 degrees apart around the circumference of the service module and the center of gravity of the spacecraft. Each propulsion unit has its own pressurization, supply, and

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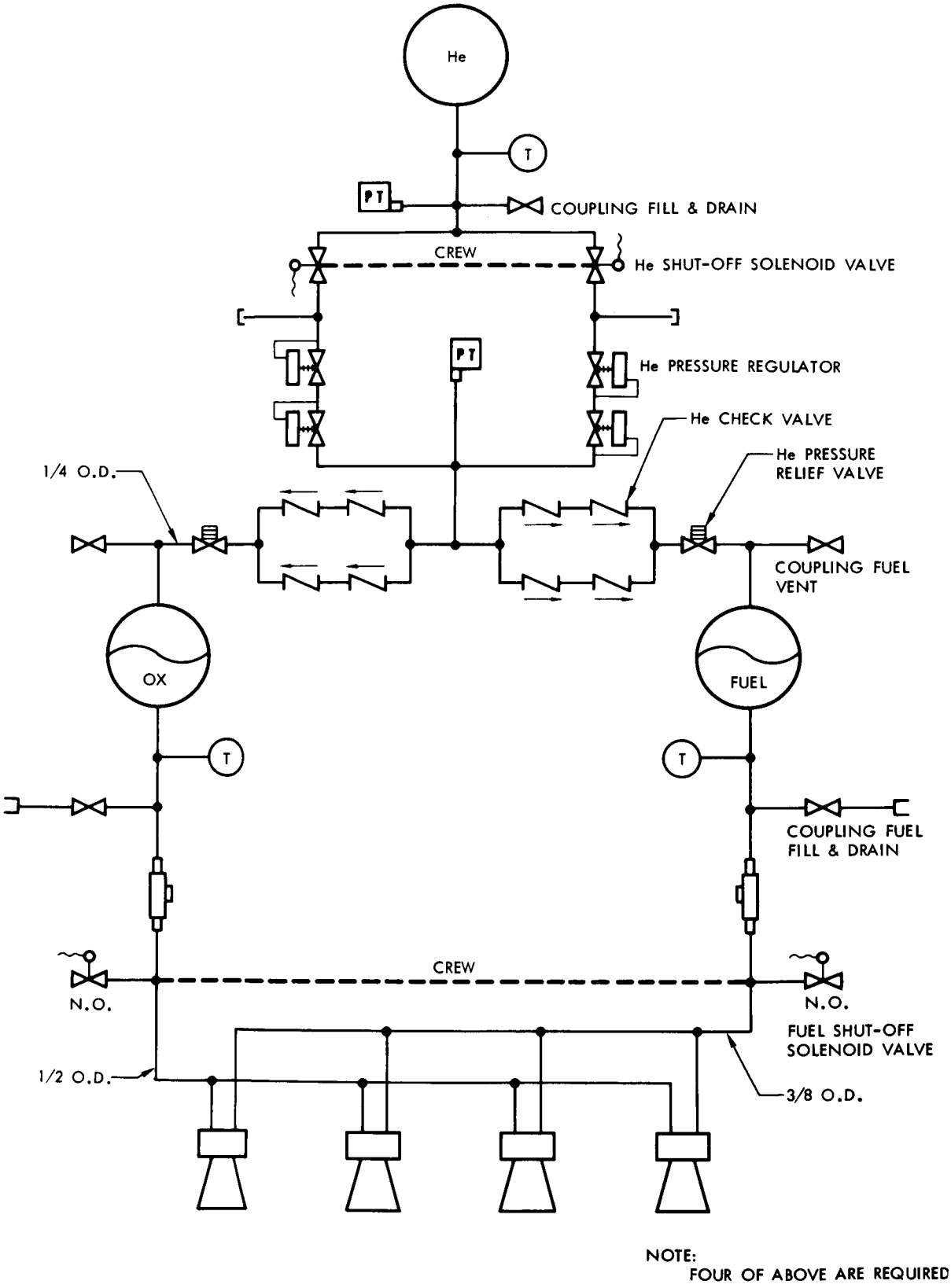

NOTE:
FOUR OF ABOVE ARE REQUIRED

Figure 3-16. Service Module Reaction Control System

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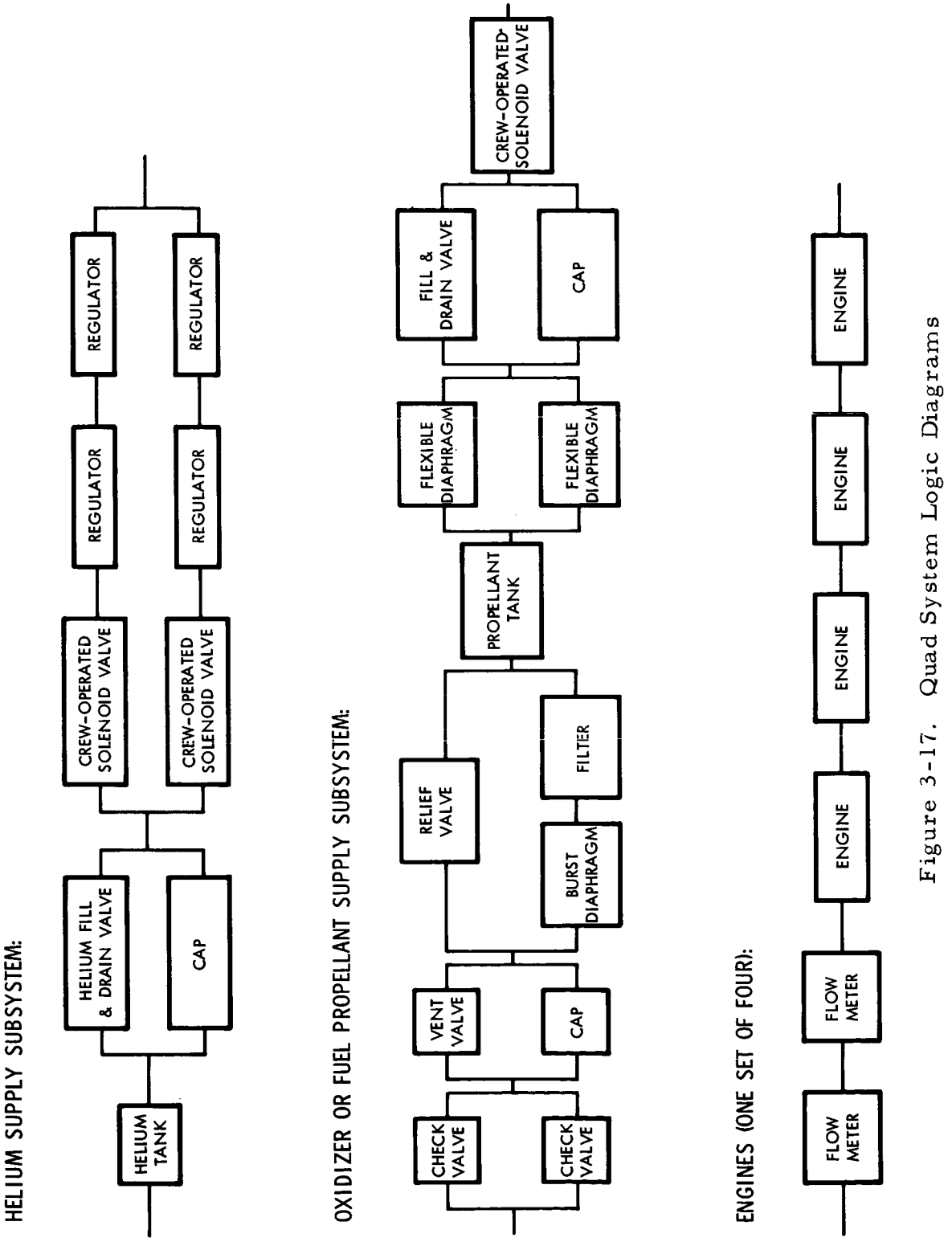

Figure 3-17. Quad System Logic Diagrams

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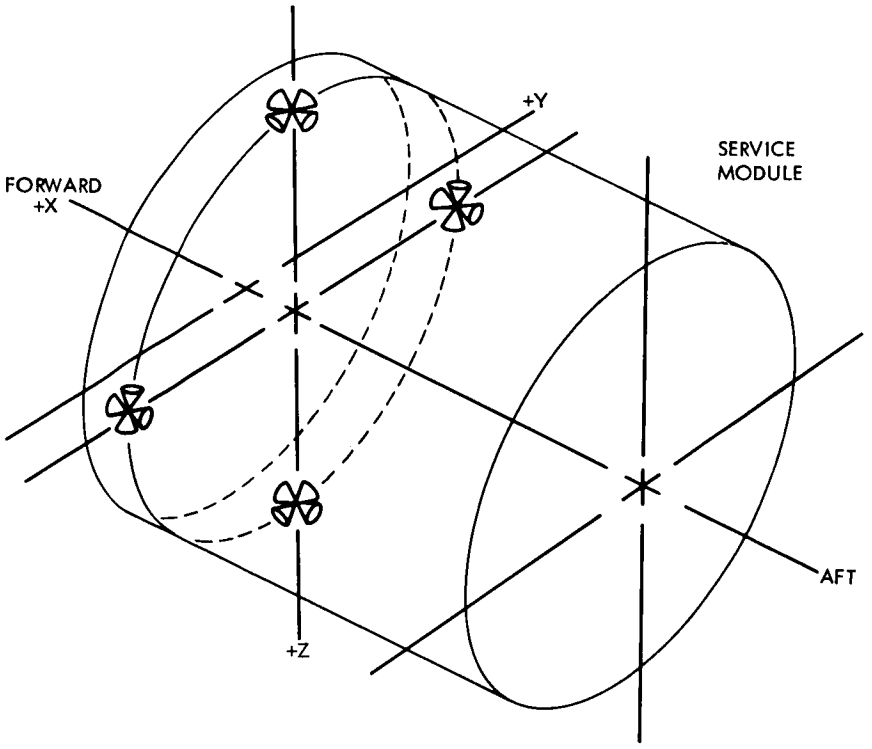


Figure 3-18. Engine Arrangement for Quad System

engine systems. The engine system in each unit is composed of four engines. Two of the engines provide control for roll maneuvers while the remaining two engines provide control for either pitch or yaw maneuvers, depending upon their location.

Several modifications of the quad system were considered for improvement and are described in the following paragraphs. The first four are major modifications, but various combinations of these modifications were investigated and evaluated. They are also described.

Modification A

Changes considered were the physical isolation of the reaction control engines to minimize the damage if an explosive engine failure occurred and the addition of two solenoid valves in each quadrant to provide a more selective engine shutdown capability. In the present quad system, if an engine valve should fail, it would be necessary to shutdown four engines, but the modified system would necessitate the shutdown of only two engines. Figure 3-19 shows the difference between the present engine configuration and the modified configuration. (See Figures 3-20 and 3-21 for additional information.)

Modification B

Modification B consists of the addition to the present quad system of four aft-pointing engines mounted at the rear of the service module. The additional engines would be supplied from the main service propulsion propellant tanks and would be operated in groups of two engines. The aft engines would provide redundancy in translation, pitch, and yaw maneuvers. It would also be possible to use the aft engines as backup for the service propulsion engine for mid-course correction and certain abort conditions. (See Figure 3-22.)

Modification C

The purpose of modification C is to provide redundancy in the translation capabilities of the SM RCS by the addition of one engine in each quadrant of the present quad system. These additional engines are mounted normal to the longitudinal thrust axis of the vehicle. (See Figures 3-23 and 3-24.)

Modification D

Modification D consists of the addition of valves and lines in order to use the gas pressure in SPS as a SM-CM separation mechanism. This

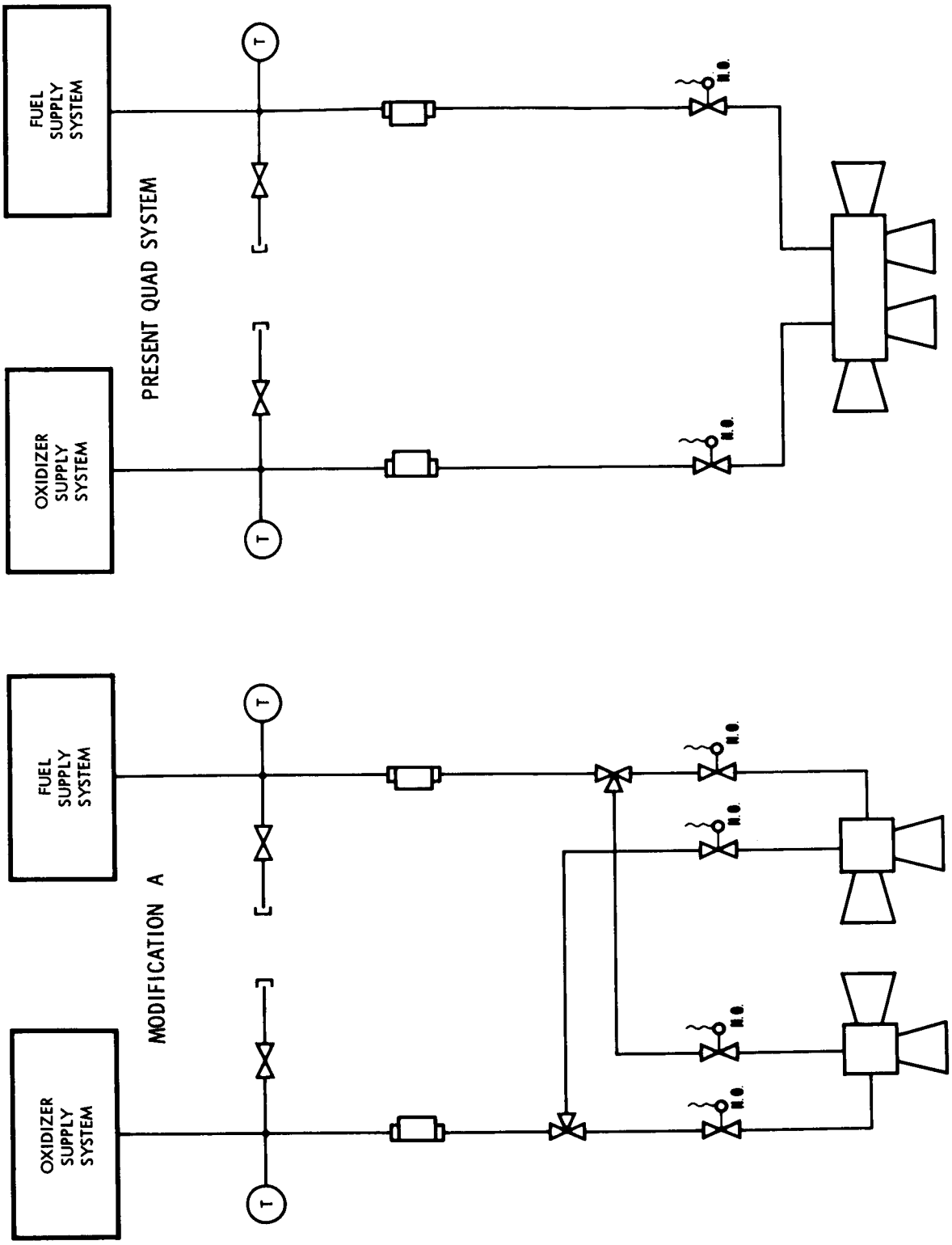
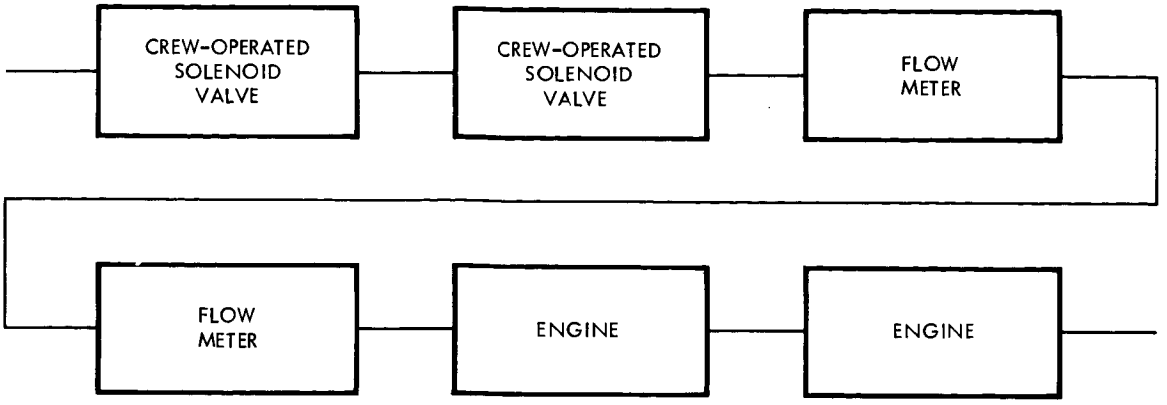


Figure 3-19. Modified and Present Quad Systems



HELIUM SUPPLY SYSTEM:
SAME AS QUAD SYSTEM
OXIDIZER OR FUEL PROPELLANT SUPPLY SUBSYSTEM:
SAME AS TRIPLE SYSTEM
ENGINES (ONE SET OF FOUR)

Figure 3-20. Modification A Logic Diagram

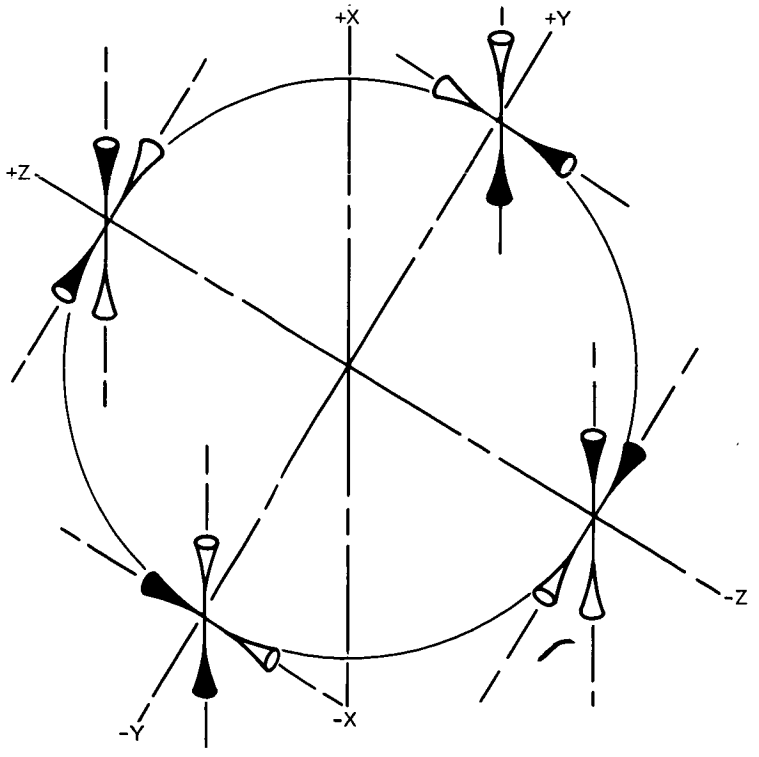


Figure 3-21. Engine Arrangement for Modification A

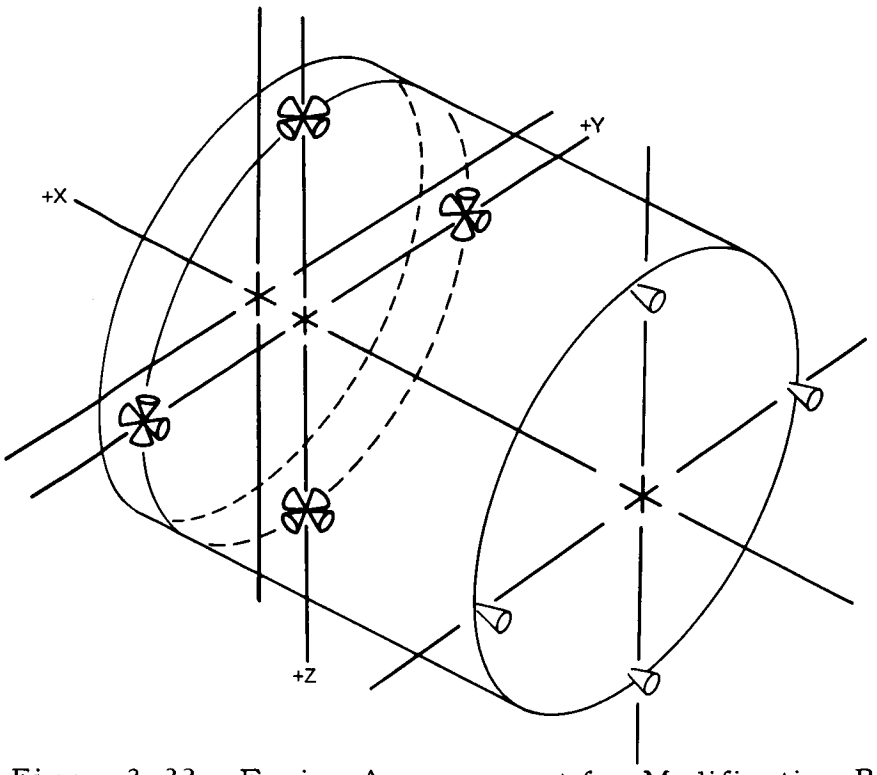


Figure 3-22. Engine Arrangement for Modification B

modification was applied to the quad system and all modified versions of the quad system. The purpose of modification D is to provide a redundant means of separation.

Modification E

Modification E is a combination of modifications A and B.

Modification F

Modification F is a combination of modifications B and C.

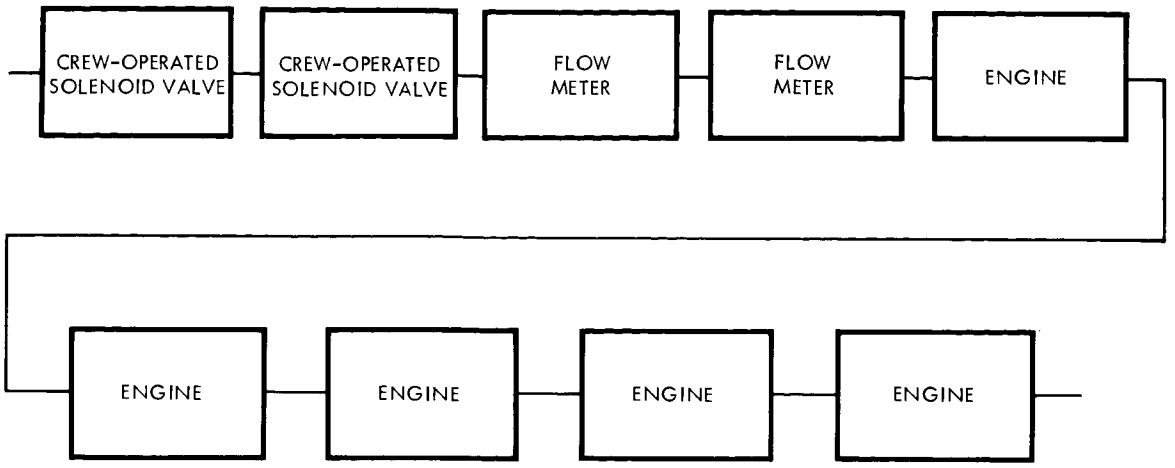
Modification G

Modification G is a combination of modifications B, C, and D.

Modification H

Modification H is a combination of modifications B and D.

The analyses of the various systems and modifications illustrated the effect of using different mission parameters and criteria. Table 3-19 shows how the reliability of the quad system varies when different mission



HELIUM SUPPLY SUBSYSTEM:
SAME AS QUAD SYSTEM

OXIDIZER OR FUEL PROPELLANT SUPPLY SYSTEM:
SAME AS QUAD SYSTEM

ENGINES (ONE SET OF FIVE):

Figure 3-23. Modification C Logic Diagram

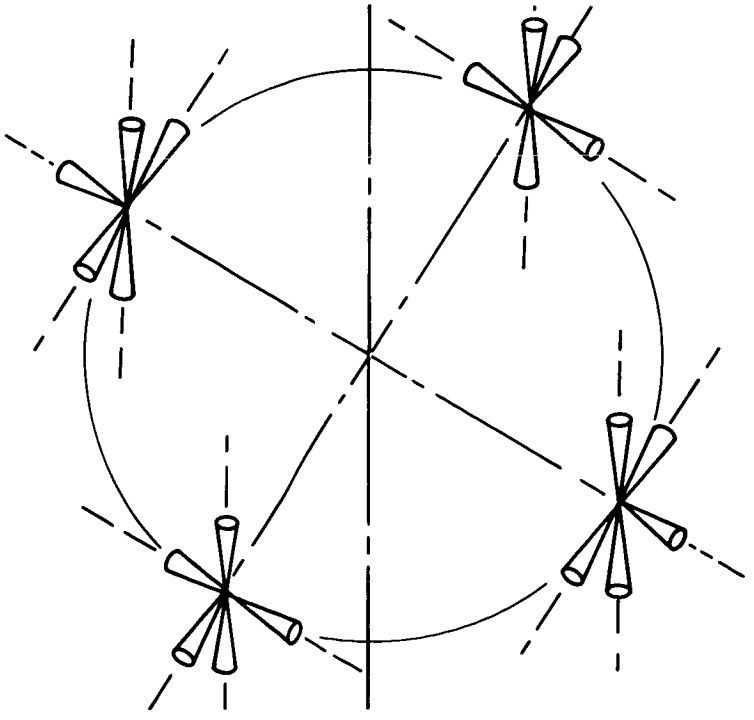


Figure 3-24. Engine Arrangement for Modification C

[redacted]

Table 3-19. Reliability Analysis Quad System

Mission Success	Crew Safety	Mission Parameters and Criteria (Assumptions)
0.958923	0.999016	1. Translation in three axes is required for the first 271 hours of the mission. 2. Minimum control is defined as requiring three out of four modules.
0.999357	0.999989	1. Three out of four modules are required for the first 271 hours of the mission. 2. Minimum control is defined as requiring any two out of four modules.
0.999123	0.999339	1. Three out of four modules are required for the first 271 hours of the mission. 2. Minimum control is defined as requiring two opposite modules for SM-CM separation.
0.973614	0.999016	1. Translation in three axes is required for the first 171 hours of the mission. 2. Minimum control is defined as requiring three out of four modules.
0.999735	0.999989	1. Three out of four modules are required for the first 171 hours of the mission. 2. Minimum control is defined as requiring any two out of four modules.
0.999260	0.999339	1. Three out of four modules are required for the first 171 hours of the mission. 2. Minimum control is defined as requiring two opposite modules for SM-CM separation.

[redacted]

Table 3-19. Reliability Analysis Quad System (Cont)

Mission Success	Crew Safety	Mission Parameters and Criteria (Assumptions)
0.998907	0.999989	1. Translation in three axes is required through LEM transposition docking. 2. Three out of four modules are required until LEM separation. 3. Translation in three axes is required from LEM separation through LEM rendezvous and docking for any combination of LEM and SM RCS. 4. Minimum control is defined as requiring any two modules.
0.998434	0.999989	1. Translation in three axes is required through LEM transposition docking. 2. Three out of four modules are required until transearth inject. 3. Any two out of four modules are required after transearth inject. 4. CM-SM separation can be accomplished with any two modules.
0.996000	0.999970	Apportioned

parameters or criteria are used. The presently accepted criteria are as follows:

1. Translation in three axes is required through LEM transposition docking.
2. Three of the four modules are required until transearth inject.
3. LEM rendezvous and docking can be accomplished with three of the four modules.
4. Minimum control requires any two of the four modules.
5. CM-SM Separation can be accomplished with a single module.

The results of the analyses are presented in Table 3-20.

Table 3-20. Reliability Quad Systems Analysis Results

Configuration	Mission Success	Crew Safety	Assumptions
Quad Modification A	0.999123 0.999075	0.999339 0.999583	1. Three out of four modules are required for the first 271 hours of the mission. 2. Opposite engines are required for SM-CM separation.
Quad Modification A	0.973614 0.973484	0.999016 0.999513	1. Translation in three axes is required for the first 171 hours of the mission. 2. Minimum control is defined as requiring three out of four modules.
Quad Modification A	0.999735 0.999675	0.999989 0.999971	1. Three out of four modules are required for the first 171 hours of the mission. 2. Minimum control is defined as requiring any two out of four modules.
Quad Modification A	0.999260 0.999288	0.999339 0.999583	1. Three out of four modules are required for the first 171 hours of the mission. 2. Opposite engines are required for SM-CM separation.
Modification B Modification E	0.998558 0.998644	0.999334 0.999603	1. Translation in three axes is required through LEM transfer only. 2. Opposite engines are required for SM-CM separation.
Modification F	0.998550	0.999006	1. Translation in three axes is required through LEM recovery. 2. Opposite engines are required for SM-CM separation.

Table 3-20. Reliability Quad Systems Analysis Results (Cont)

Configuration	Mission Success	Crew Safety	Assumptions
Modification G	0.999598	0.999982	1. Translation in three axes is required through LEM recovery. 2. SM-CM separation can be accomplished with helium gas.
Modification H	0.999157	0.999980	1. Translation in three axes is required through LEM transfer. 2. SM-CM separation can be accomplished with helium gas.
Modification H	0.998580	0.999991	1. Translation in three axes is required through LEM transposition docking. 2. Three out of four quadrants are required until LEM separation. 3. Rotation in three axes is required throughout the flight. 4. Translation in three axes is required for LEM separation through LEM rendezvous and docking for any combination of LEM and SM. 5. SM-RCS aft engines are used after one failure. 6. SM-CM separation can be accomplished with helium gas.

Table 3-20. Reliability Quad Systems Analysis Results (Continued)

Configuration	Mission Success	Crew Safety	Assumptions
Modification H	0.987363	0.999983	1. Translation in three axes is required through lunar orbit inject. 2. Should the remaining equipment provide only minimum control prior to LEM separation, the mission will be aborted. 3. The aft engines are used only after two modules have failed. 4. Minimum control is required from LEM separation through SM-CM separation. 5. SM-CM separation can be accomplished with helium gas pressure.
Modification H	0.998392	0.999991	1. Translation in three axes is required through LEM transfer only. 2. Three out of four modules are required through lunar orbit inject. 3. The aft engines are used after one quadrant failure. 4. Should the remaining equipment provide only minimum control prior to LEM separation, the mission will be aborted. 5. SM-CM separation can be accomplished with helium gas pressure.

Table 3-20. Reliability Quad Systems Analysis Results (Cont)

Configuration	Mission Success	Crew Safety	Assumptions
Quad	0.998907	0.999989	1. Translation in three axes is required through LEM transposition docking. 2. Three out of four modules are required until separation. 3. Translation in three axes is required from LEM separation through LEM rendezvous and docking for any combination of LEM and SM-RCS. 4. Minimum control is defined as requiring any two modules.
Apportioned	0.996000	0.999970	

Apollo Reliability Engineering will continue to perform studies leading to maximum reliability improvements.

Problem Areas

The lack of exacting definitions for mission success and crew safety has been a major handicap necessitating numerous analyses and studies dealing with the effect that varying mission parameters would have upon the reliability of the system. The major mission requirements allowed to vary in the analyses were as follows:

LEM transposition docking
Transearth mission success
Lunar orbit prior to LEM separation
LEM rendezvous and docking
Propellant settling maneuver
CM-SM separation
Minimum control definition

TEST PROGRAM

Qualification Test

Information was supplied for inclusion in Volume III (Ground Qualification Tests) of SID 62-109. The data included a definition of the systems test program, a system logic diagram, a presentation of system criticality versus hardware requirements, and a qualification test schedule with test specimen utilization.

Engine qualification tests are scheduled to commence in October 1963. Component qualification tests are scheduled to commence in May or June 1963.

Development Tests

The Marquardt development test program is approximately thirty-percent complete. The component development test phase is approximately five percent complete on the basis of subsystem development and material evaluation tests.

At the present time, three engines are being used in the program. One engine was fired for 4381 seconds at altitude with no deleterious effects. This engine has also undergone vibration testing, during which valve testing was noted at the high g levels. Another engine was submitted to a 10-day humidity soak. All requirements were satisfied, but a slight leakage was observed. Both of these occurrences are being investigated. The

propellant valves were tested for short-pulse performance as well as high-temperature exposure. At the conclusion of the latter test, 200,000 cycles were performed with no seat leakage. Material tests are continuing on the combustion chamber coating.

Development tests on the engine will continue, including steady-state, short-pulse, mission-duty, and off-nominal firing conditions. Environmental tests will include vibration, shock, low temperature, and fungus.

Specific component development tests have not been started, but all such tests will begin in the next quarter.

SUBCONTRACTOR MANAGEMENT

Malfunction of two oxidizer valves, caused by contamination of that specified in MA 0610-004 is being investigated thoroughly. Marquardt Reliability to determine proper corrective action. A report will be issued at the conclusion of the investigation.

The Marquardt documents that will be completed and presented to S&ID during the next quarter are as follows:

Reliability Program Plan (Revised)
Failure Mode Analysis and Apportionments (Revised)
Injector Head Assembly Design Review
Thrust Chamber Design Review

Studies have been initiated by Marquardt Reliability to develop a means for estimating reliability from truncated test results of equipment subject to wear-out failures. This study is approximately fifty-percent complete. Statistical procedures for handling variabilities of measurement instruments also have been started.

PLANNED ACTIVITIES

The SM RCS will be subjected to an updated failure mode and effects analysis, and the results will be presented in the next quarterly progress report.

A study will be conducted on the instrumentation requirements for the SM RCS. The purpose of the investigation will be to determine the amount and type of instrumentation needed to detect critical failures within the system to improve the possibility of a successful mission.

A crew safety design review will be conducted for the SM RCS to assure that all significant factors affecting reliability and crew safety have been evaluated and that recommended action for the factors that may degrade inherent reliability has been accomplished.

SERVICE PROPULSION

SUMMARY

The Reliability Engineering effort for the past quarter was expended in the reevaluation of the service propulsion system equipment operating modes at the detail part level. This resulted in significant changes from previously submitted logic diagrams. For example, all components that are nonoperative or unpressurized during the coast phase were assigned a passive failure rate, and the helium solenoid valve will be closed during all coast periods. Results of the worst-case analysis indicate a mission success reliability that meets the SPS apportionment of 0.999770.

Mission success and crew safety reliability is determined as a function of the established abort criteria, which states that the mission will be aborted if sufficient failures occur in operating systems or equipment so that one additional failure will eliminate the capability of safe abort. In individual cases, where an unacceptable tradeoff between mission success and crew safety reliability will be produced, these criteria may be modified. Figures 3-25 through 3-29, located at the end of the section, illustrate logic diagrams of the SPS and indicate the redundancy available. Though the detailed components are designed to minimize failure rates, failures will result in mission abort instances. The probability of failures is lessened because required operating times are short.

A propellant utilization system optimization approach was developed for Engineering to assist in selection of the propellant utilization system supplier.

Based on the failure mode analysis, it was evident that available failure rate data were insufficient to justify designating the superiority of either the capacitance or the point sensor systems. Using the criteria that redundant systems should not be subject to identical failure modes, the combination of these two systems does represent the most reliable application of state-of-the-art reliability and design philosophy.

Spacecraft leakage during fabrication must be detected at lower pressures than planned because of safety codes. An analysis was conducted for the effect of reduced pressure testing on system reliability.

It is recommended that the spacecraft propellant system components be assembled to the highest possible level before assembly in the spacecraft.

This will provide leak testing at a high level of assembly at working pressure and will keep the number of joints that must be checked in the spacecraft to a minimum.

The current logic diagrams and reliability equations are taken from the failure mode analysis. The significant changes in the logic include the following assumptions.

The Helium solenoid valve's will be closed during all coast periods. The Engine subsystem propellant valves will be pressurized during all coast periods. All components which are non-operative or unpressurized during coast period are shown at a passive failure rate. All components leakage redundancy is evaluated during coast. This includes internal part redundancy in components. This criteria produces a service propulsion system reliability of 0.999772 using original system apportionments.

ANALYSIS

Mission Success

Mission success is determined by considering the service propulsion system to be in the mission success mode until transearth injection and in the crew safety mode until service module separation. Figure 3-25 shows the mission success mode of the helium and propellant supply systems. The difference between the diagrams illustrated in Figures 3-25 and 3-26 is the arrangement of the solenoid valves and regulators, which are in series for mission success and redundant in the crew safety mode. Figure 3-27 shows the mission success logic for the engine system with major components arranged in series

Mission success calculations are based on all components operating. The probability of mission successes are classified by the following phases.

P_s = Propellant supply system exclusive of solenoid and regulators

$P_{SR(passive)}$ = propellant supply systems passive with passive failure rates for regulators not required to work during coast periods

$P_{E(passive)}$ = Engine system passive with passive failure rates used for all components not required to work under stress for coast periods

$P_{E(operational)}$ = Engine system operational with all components at mission apportionment during firing

And combined by the independent relationship:

$$P_{ms} = P_s \times P_{SR(passive)} \times P_{SR(operational)} \times P_{E(passive)} \times P_{E(operational)}$$

Crew Safety

Figures 3-26 through 3-29 indicate the redundant modes for crew safety failure probabilities. The helium and propellant supply systems in the redundant mode are shown in Figure 3-26. The solenoid valves and regulators are in parallel for the crew safety mode.

Figure 3-28 is the logic diagram for the engine system in the passive phase. In this logic diagram the main pilot solenoid valves are considered to be redundant for leakage failure and in series arrangement for passive failure. As a result, it will be recommended to the engine supplier that the main pilot solenoid valves have redundant seals.

Figure 3-29 is the logic diagram for the redundant engine system. The propellant feed valves and actuators and the gimbal actuators are shown as redundant for crew safety operation. The components are redundant for crew safety for the entire mission.

SPECIAL STUDIES

Propellant Utilization System

Reliability criteria were established for determining an optimum approach for the propellant utilization system. Failure mode considerations best define the design redundancy requirements. They should not affect the primary and secondary systems to the same degree. Failure modes for this system are the deterioration of propellants causing an apparent off-ratio, open or short circuits. These failure modes should be minimized and, where the environment is critical, reliability design should preclude identical failure modes. Either system should be capable of checking the other. Single equipment failures should not result in loss of the entire system.

Available failure rate data is insufficient to make a selection between the capacitance or point sensors. Neither system design alone can provide both the redundancy and the operating capabilities for mission success. The combination of these two systems represent the most reliable application of state-of-the-art reliability and design philosophy and is recommended by Reliability as the most desirable system for use in the service module-SPS.

System Checkout Leakage Detection

The preliminary system checkout specification for leakage detection was reviewed. Reliability requirements can be assured only by adequate testing during assembly. Since the individual components are leak tested at working pressure, the subassemblies require testing to the same level in order to meet the allowable mission failure rate for mechanical fittings and tubing. Using the most optimistic current state-of-the-art failure rates for fittings and tubing (0.04×10^{-6}), a system comprised of 100 joints would contribute 1344 failure per 10^6 missions. This is greater than the apportioned reliability of the entire system, and neither crew safety nor mission success could be met using this criteria. Information available from the B-70 program for hazardous fittings established the following criteria: The fittings must be designed for a strength to weight ratio of 3.5 to 1 and must have 100 percent inspection to detect faulty or damaged joints. Based on these parameters, the failure rate is 26 per 10^6 missions for a system comprised of 100 joints.

It is reliability's recommendation that the spacecraft propellant system components be assembled to the highest possible level before assembly into the spacecraft. This will provide leak testing at a high level of assembly at working pressure and will keep the number of joints that must be checked in the spacecraft to a minimum. If test facilities are not available and plant safety will not allow testing the final spacecraft at working pressure, then the fitting and joint program should contain adequate testing to allow statistical evaluation of joint testing at lower pressure levels. This would require more stringent leakage requirements, which would be in agreement with the viscous-flow concept described in Consolidated Electrodynamics Corporation document, Aspects of Leak Detecting Hermetic Seals Using a Mass Spectrometer Type Leak Detector. This document states that leakage increases as the square of the difference in pressure ratio.

Problem Areas

Two major problem areas within the SM service propulsion system that have become apparent are the inability of the present configuration to meet crew safety requirements and the increasing difficulty of expecting engine reliability improvement commensurate with Apollo requirements. The first was caused primarily by the change to the LOR operational concept, and the second is the inability of the subcontractor to achieve performance goals.

Evaluation of the present service propulsion system configuration has resulted in predicted crew safety of 0.999800, as compared to the apportioned value of 0.999970. Reliability analysis has shown that small component redundancy has been optimized and that component improvement has been

extended as much as the schedule will allow. Additional analytical work on this configuration has little chance of providing a significant improvement in crew safety. Principal reasons for the low system reliability are the non-redundant components, such as helium tanks, propellant tanks, and rocket engine. Several areas where redundancy may be incorporated will be studied to devise a system of reasonable weight and higher crew safety. Studies will include LEM rocket engine back-up operational modes, and reduction of number of components under continuous stress.

The thrust chamber assembly has little possibility of attaining the predicted 0.999990 within the required time schedule, since present subcontractor tests have not been within specifications. No demonstration of specification performance has been made to date. Either the specific impulse has been low or chamber erosion has been excessive. Combustion instability has been present persistently throughout the development program. Solutions to these problems are expensive and time consuming. Other rocket programs have taken several years to increase engine reliability to 0.995000. At the inception of the Apollo program, Aerojet-General expressed confidence in an injector design based on Titan II tests and in a chamber design based on a previous Air Force program. Initial designs have not met with the specification goals, since testing has shown no successes for the year of engine contract. A re-evaluation of the single engine concept is necessary at this time.

TEST PROGRAM

Qualification Test

Information was submitted for Vol. III (Ground Qualification Tests) of the General Test Plan Research and Development for Apollo Spacecraft (SID 62-109). The data includes a definition of the systems test program, a system logic diagram, a presentation of system criticality versus hardware requirements, and a qualification test schedule with test specimen utilization.

General Test Program Status

Percent Complete on Development Tests

The Aerojet Development Test program is approximately 55 percent complete. Development test phase is approximately 5 percent complete on the basis of system development and material evaluation tests.

Qualification Status

Component Qualification tests are scheduled to begin in May or June 1963.

Specific Development Tests

The subscale test program has been virtually completed. The results of these tests are being processed. C* Verification tests and injector pattern evaluation tests are being conducted at Sacramento on various combinations of baffled and unbaffled injectors. Material tests are continuing on the bearing lubricant and the ablative material. In the next quarter, development tests will continue on the engine, propellant valves, and gimbal actuator. Specific component development tests have not begun but these tests will commence in the succeeding quarter.

SUBCONTRACTOR MANAGEMENT

The impact of the Aeroject-General move from Azusa to Sacramento has not been fully evaluated. The intention is to transfer key reliability personnel so that continuity of the reliability program can be maintained. There is at present no firm indication of the number of people accepting the transfer. In anticipation of the move, no increase in reliability personnel has occurred in the last few months. This under-staffing has resulted in slippage of the design review schedule and the failure analyses.

Failure reporting procedure has been clarified. Documentation specification MC999-0025 called for special reporting of significant failures which has now been defined as any test discrepancy outside of Procurement Specification MC901-0009A limits. During qualification, all significant failures shall be reported to a central file at Aerojet-General Corporation. Any failures which result in a schedule or design change shall be reported to NAA/S&ID. Failure reporting to date has been verbal only. Written notification and analyses are in process.

PLANNED ACTIVITIES

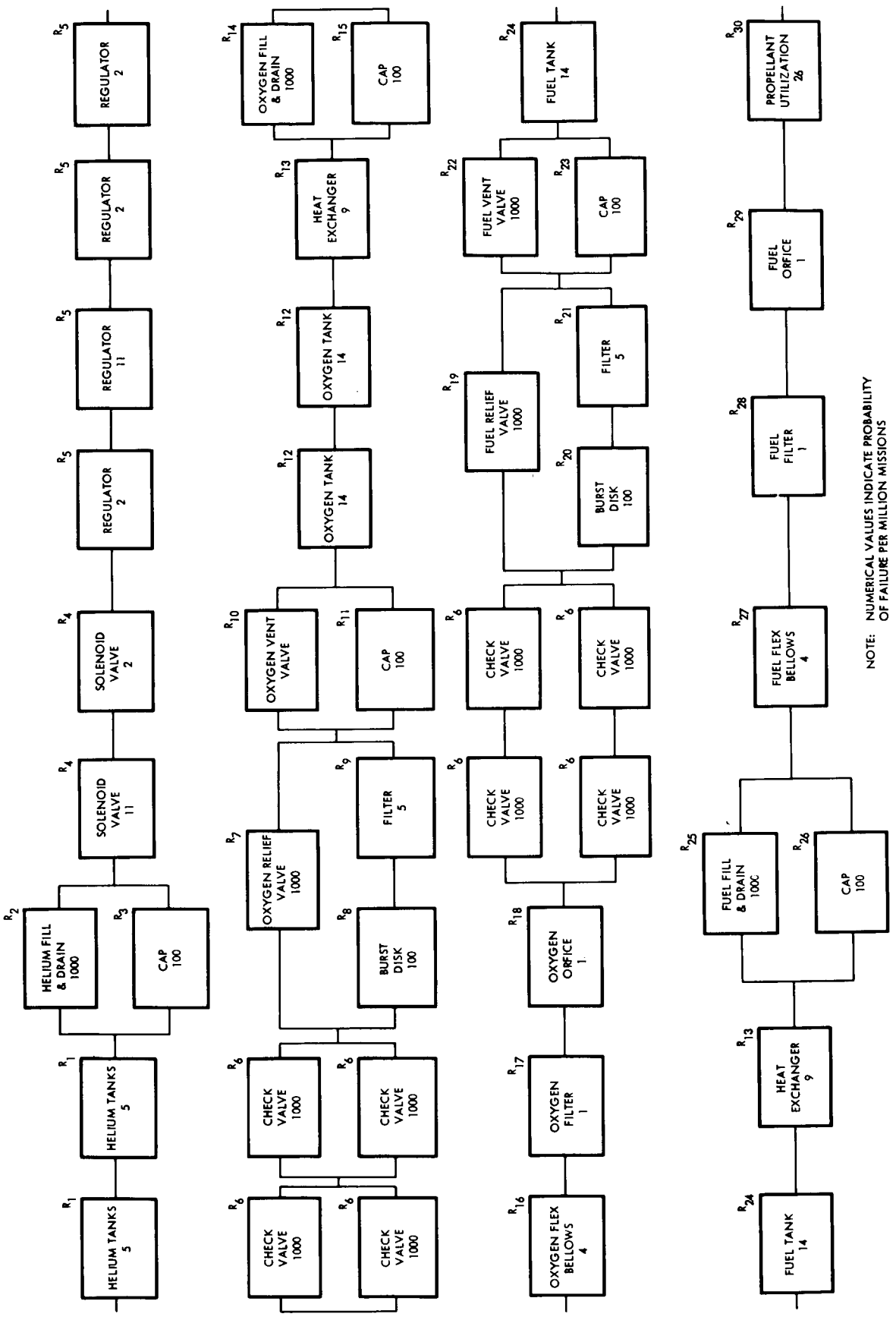
The crew safety requirements cannot be obtained by use of additional redundancy of smaller components in the service propulsion system. A study is planned using the helium and propellant supply systems in parallel for portions of the mission. A study encompassing possible alternate back-up modes for the service propulsion system is being conducted to explore all possible methods of enhancing service propulsion system crew safety. Configuration changes required to incorporate this will also be

[redacted]

included. A preliminary reliability crew safety design review for the service propulsion systems is scheduled for the beginning of April.

Negotiations with tentative suppliers of fuel and oxidizer quick disconnects are in progress. The reliability requirements are being reviewed, and agreement will be reached on the scope of the individual programs.

As other suppliers are selected for service propulsion system components, the reliability requirements will be coordinated with the vendor and firm programs established.



$$P_s = R_1^2\left[1-(1-R_2)(1-R_3)\right] R_4^2 R_5^4 \left[1-(1-R_6^2)^2\right]^2 \left[1-(1-R_8^2)(1-R_9)\right] \left[1-(1-R_{10})(1-R_{11})\right] R_{12}^2 R_{13}^2 \left[1-(1-R_{14})(1-R_{15})\right] R_{16} R_{17} R_{18} \left[1-(1-R_{19})(1-R_{20}R_{21})\right] \left[1-(1-R_{22})(1-R_{23})\right] R_{24}^2 \left[1-(1-R_{25})(1-R_{26})\right] R_{27} R_{28} R_{29} R_{30}$$

Figure 3-25. Service Module Service Propulsion System—Mission Success Helium and Propellant Supply Systems Logic Diagram

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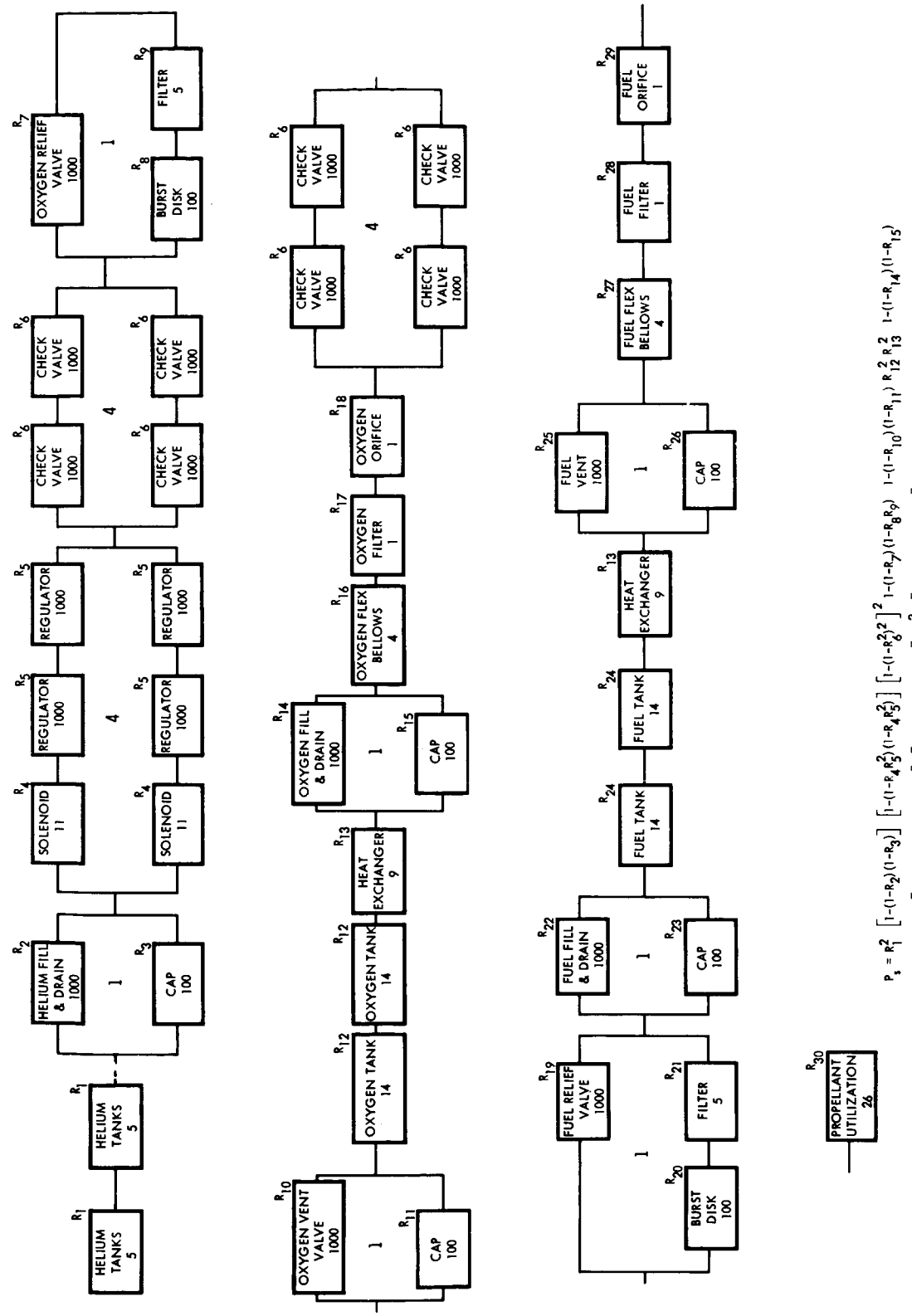

$$P_s = R_1^2 \left[1-(1-R_2)(1-R_3)\right] \left[1-(1-R_4R_5^2)(1-R_4R_5^2)\right] \left[1-(1-R_8^2)^2\right]^2 1-(1-R_7)(1-R_8R_9) \quad 1-(1-R_{10})(1-R_{11}) R_{12}^2 R_{13}^2 \quad 1-(1-R_{14})(1-R_{15})$$
$$R_{16} R_{17} R_{18} \left[1-(1-R_{19})(1-R_{20} R_{21})\right] \left[1-(1-R_{22})(1-R_{23})\right] R_{24}^2 \left[1-(1-R_{25})(1-R_{26})\right] R_{27} R_{28} R_{29} R_{30}$$

NOTE: NUMERICAL VALUES INDICATE PROBABILITY OF FAILURE PER MILLION MISSIONS

Figure 3-26. Service Propulsion System—Crew Safety Helium and Propellant Supply Systems Logic Diagram

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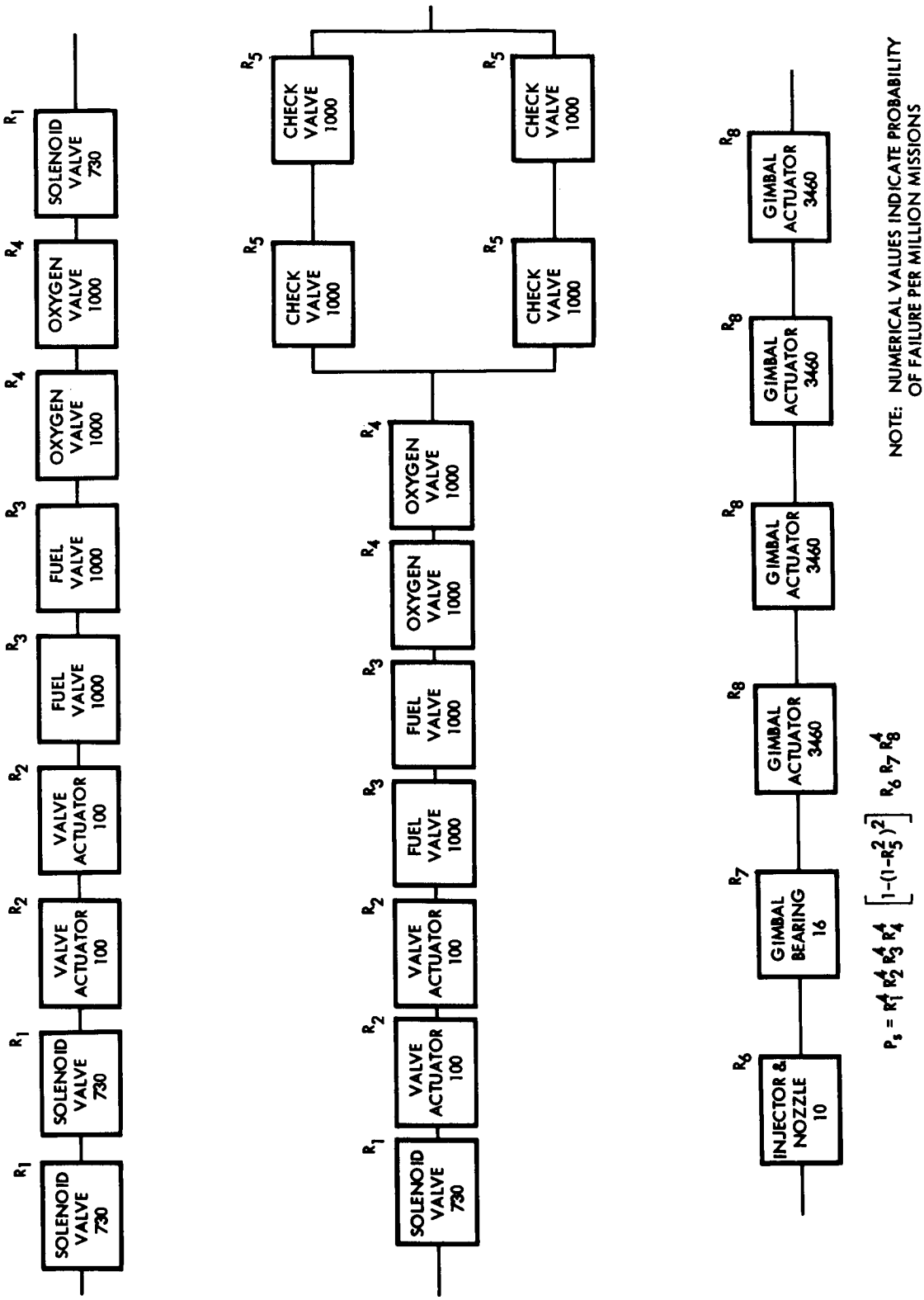
R1 SOLENOID VALVE 730
R4 OXYGEN VALVE 1000
R4 OXYGEN VALVE 1000
R3 FUEL VALVE 1000
R3 FUEL VALVE 1000
R2 VALVE ACTUATOR 100
R2 VALVE ACTUATOR 100
R1 SOLENOID VALVE 730
R1 SOLENOID VALVE 730
R2 VALVE ACTUATOR 100
R2 VALVE ACTUATOR 100
R3 FUEL VALVE 1000
R3 FUEL VALVE 1000
R4 OXYGEN VALVE 1000
R4 OXYGEN VALVE 1000
R5 CHECK VALVE 1000
R5 CHECK VALVE 1000
R5 CHECK VALVE 1000
R5 CHECK VALVE 1000
R6 INJECTOR & NOZZLE 10
R7 GIMBAL BEARING 16
R8 GIMBAL ACTUATOR 3460
R8 GIMBAL ACTUATOR 3460
R8 GIMBAL ACTUATOR 3460
R8 GIMBAL ACTUATOR 3460
$P_s = R_1^4 R_2^4 R_3^4 R_4^4 \left[1-(1-R_5^2)^2\right] R_6 R_7 R_8^4$
NOTE: NUMERICAL VALUES INDICATE PROBABILITY OF FAILURE PER MILLION MISSIONS

Figure 3-27. Service Propulsion System—Engine System Mission Success Logic Diagram

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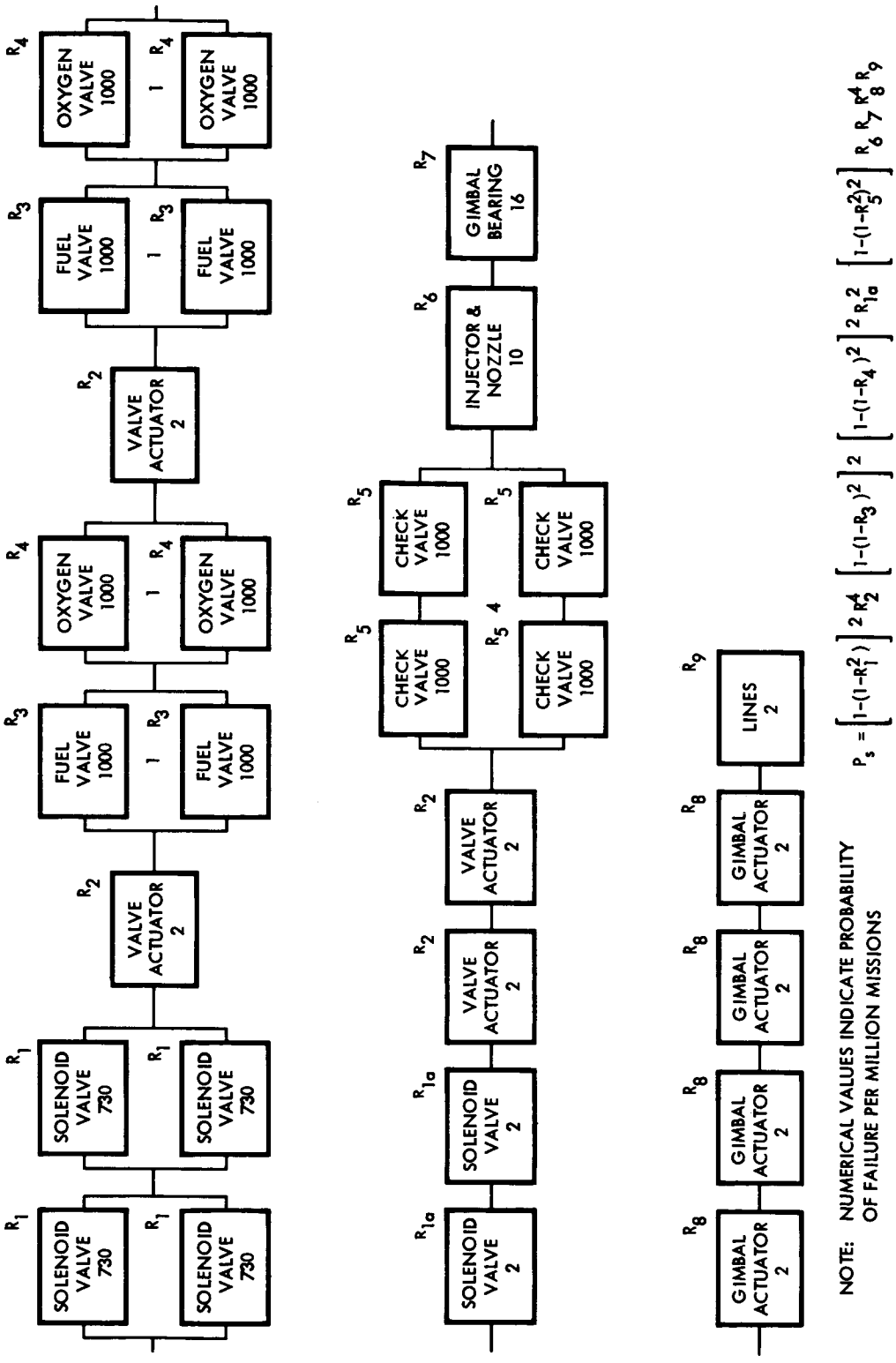
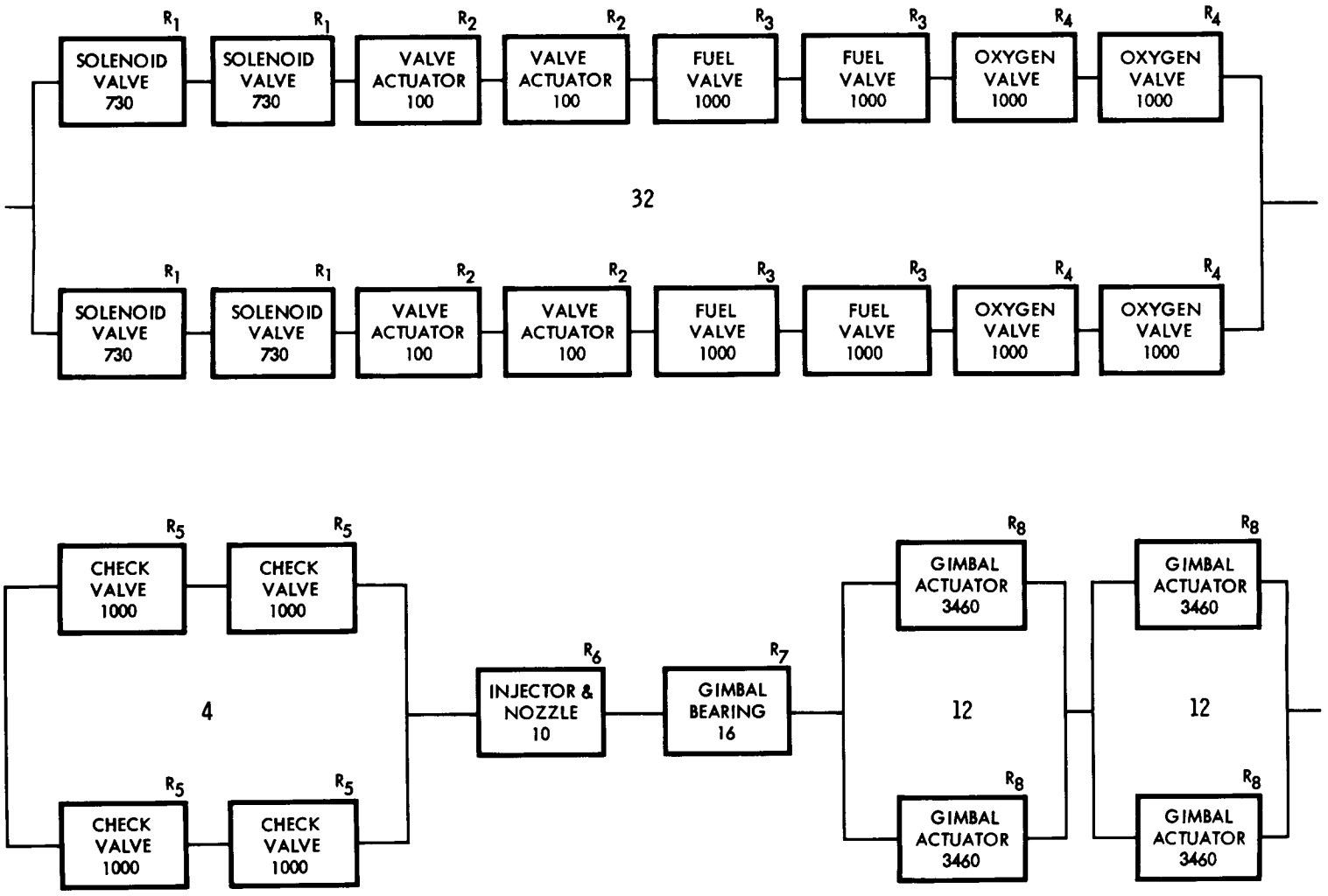

R1 SOLENOID VALVE 730
R1 SOLENOID VALVE 730
R1 SOLENOID VALVE 730
R1 SOLENOID VALVE 730
R2 VALVE ACTUATOR 2
R3 FUEL VALVE 1000
R3 FUEL VALVE 1000
R4 OXYGEN VALVE 1000
R4 OXYGEN VALVE 1000
R2 VALVE ACTUATOR 2
R3 FUEL VALVE 1000
R3 FUEL VALVE 1000
R4 OXYGEN VALVE 1000
R4 OXYGEN VALVE 1000
R1a SOLENOID VALVE 2
R1a SOLENOID VALVE 2
R2 VALVE ACTUATOR 2
R2 VALVE ACTUATOR 2
R5 CHECK VALVE 1000
R5 CHECK VALVE 1000
R5 CHECK VALVE 1000
R5 CHECK VALVE 1000
R6 INJECTOR & NOZZLE 10
R7 GIMBAL BEARING 16
R8 GIMBAL ACTUATOR 2
R8 GIMBAL ACTUATOR 2
R8 GIMBAL ACTUATOR 2
R8 GIMBAL ACTUATOR 2
R9 LINES 2
NOTE: NUMERICAL VALUES INDICATE PROBABILITY OF FAILURE PER MILLION MISSIONS
$P_s = \left[1-(1-R_1^2)\right]^2 R_2^4 \left[1-(1-R_3)^2\right]^2 \left[1-(1-R_4)^2\right]^2 R_{1a}^2 \left[1-(1-R_5^2)^2\right] R_6 R_7 R_8^4 R_9$

Figure 3-28. Service Propulsion System—Engine System Passive Stage Logic Diagram



NOTE: NUMERICAL VALUES INDICATE PROBABILITY OF FAILURE PER MILLION MISSIONS

$$P_s = \left[1-(1-R_1^2 R_2^2 R_3^2 R_4^2)\right] \left[1-(1-R_5^2)^2\right] R_6 R_7 \left[1-(1-R_8)^2\right]^2$$

Figure 3-29. Service Propulsion System—Engine System Crew Safety Logic Diagram

COMMAND MODULE HEAT SHIELD

SUMMARY

The major reliability effort for this period has been directed toward further developing of analytical techniques for reliability assessment, preparing revised qualification test procedures, and reviewing of environmental criteria.

The method of reliability assessment was described in the Fourth Quarterly Reliability Status Report, SID 62-557-4. A quantitative reliability assessment will not be possible until the subcontractor's (AVCO-RAD) test program develops heat shield back-face temperature data from testing samples exposed to critical environmental conditions. Samples will be representative of the current filled honeycomb material. The best estimate of reliability will be included, when available. Until that time, it is assumed that the heat shield ablative panels will attain the reliability goal of 0.99995.

ANALYSIS

A statistical regression analysis was performed on six sets of Apollo ablative material test specimens to derive heat of ablation (q*) versus air enthalpy (H/RT_o) measurements. The objective of the analysis was to compare and evaluate the ablative performance of the experimental materials.

Statistical analyses were performed on the mechanical properties of Avcoat 5026-39 filled, 3/4-inch sinusoidal HRP honeycomb material. These properties were derived from compression tests on cylindrical and rectangular specimen configurations. The objective of the analyses was to derive definitive information to be used for determining whether cylindrical or rectangular-shaped core specimens should be cut from test panels.

Failure Effects Analysis

The failure mode logic network presented in Figure 3-30 delineates an orderly approach for determining and analyzing the potential failure types and their effect on the Apollo heat shield. This approach ensures that all important items or design areas of the ablative (Avcoat 5029-39) filled, fiberglass honeycomb bonded to the substructure have been analyzed and the

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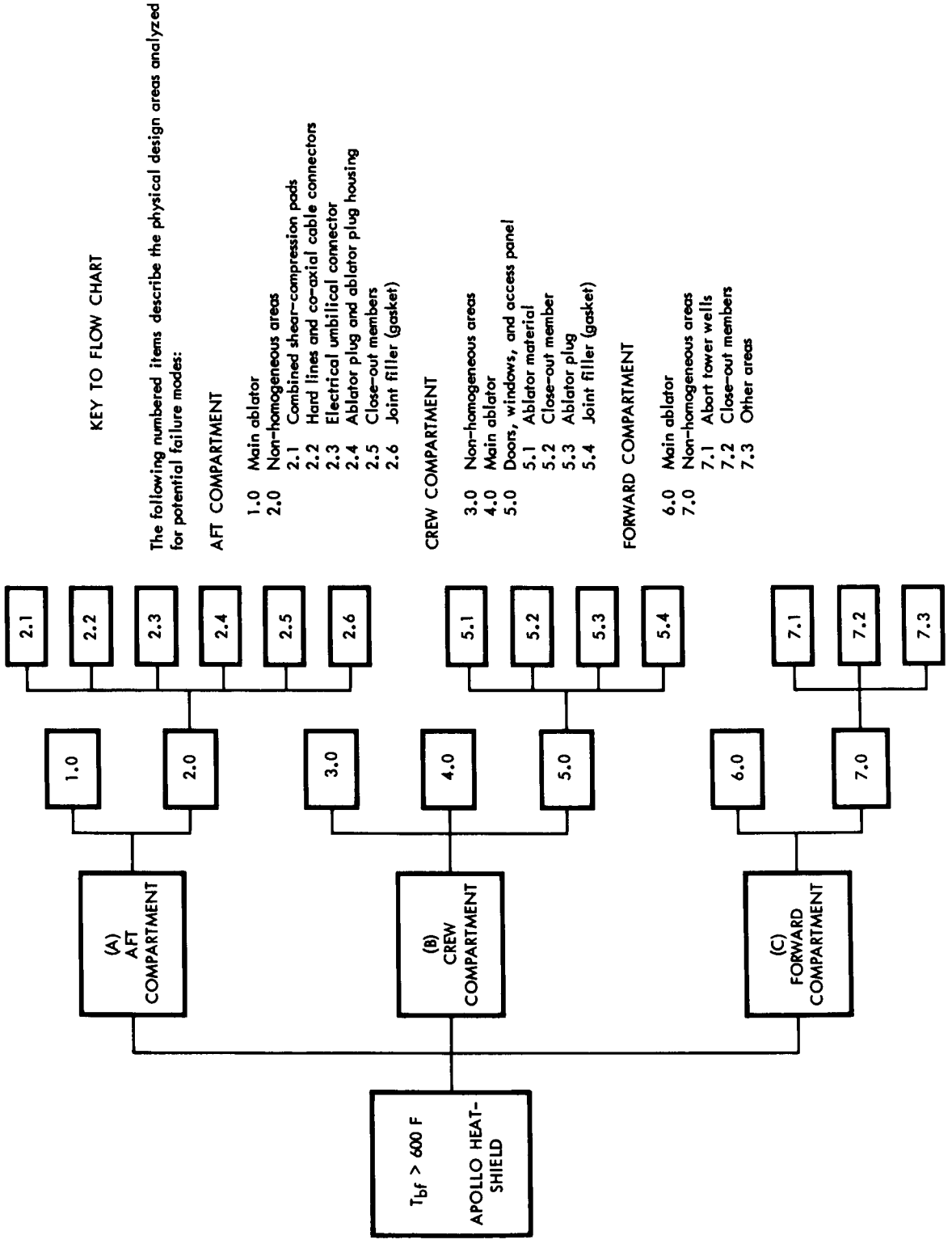

Figure 3-30. Failure-Type Logic Network

results recorded in the failure effects analysis shown in Table 3-21. The back-face temperature (T_{bf}) is the criteria for determining a heat shield failure, as specified in NAA/S&ID Procurement Specification, MC 364-0001.

Nondestructive Testing Techniques

The nondestructive, bond strength studies have been delayed because of the honeycomb material design change. Tests will begin during the next quarter after new sample specimens have been tested and bonded to the test fixture.

An investigation of existing nondestructive testing techniques indicated a general difficulty in attaining adequate, high-reliability measurements. Therefore, strong support was given to the general requirements for adhesive bonding specification, now in preparation. Acceptance and adoption of this general guide for all future bonding specifications was recommended.

A recommended matrix design formulated for the sequential (ablation) tests was prepared. This matrix design is being modified to reflect the deletion of the electron/proton radiation portion of the test.

Data Collection, Storage, and Analysis

Data on the effects of atmospheric environments on the materials and components have been collected and abstracted for compilation and dissemination to design and reliability engineers.

The interservice data exchange program (IDEP) and other out-of-house data sources are continuously monitored by data control personnel to provide the most current data generated by other contractors in similar fields of research. Abstracts of pertinent reports have been routed to reliability personnel.

Continuous liaison has been maintained with in-house environmental laboratory personnel to ensure that appropriate test data are stored to provide backup information for the reliability assessment program. Modification to current mechanized storage methods have been under study to provide the capability for more rapid and frequent scanning and print-out of such accumulated data to permit greater utilization of available data.

Development test data have been organized and prepared for statistical reduction and analysis. These data consist of the results of measurements performed during mechanical, thermal, chemical, and physical properties tests on heat shield ablative material. Development test data will be used, as a supplement to qualification test data, for the establishment of the best estimate of heat shield reliability. This assessment technique will use

Table 3-21. Failure Effects Analysis

Item	Drawing Number	Function	Failure Type	Mission Phase	Cause	Failure Effect			Posible Crew Action
						On Heat Shield Performance	On Subsystem Performance	On Apollo	
Aft Compartment 1.0 Main ablator	LA-4583	Thermal protection; honeycomb provides structural rigidity to ablator	1.0.1 Discontinuous separations between honeycomb (HC) and ablator pencils	(a) Lift off, boost, earth orbit, trans-lunar, lunar orbit, transearth (b) Atmospheric entry	In-plane tension	(a) Minor (b) Local heat penetration, intensity depending on time in entry phase	(a) Minor (b) Minor to severe, depending on temperature extent and location	(a) Minor (b) Minor to catastrophic	If known, external in-flight repair of toroidal zone only; an abort mode from, or after a return to, if possible, an earth orbit: or no action
			1.0.2 Discontinuous separation between HC ribbons at node location	Same as 1.0.1(a), (b)	In-plane tension	Same as 1.0.1(a), (b)	Same as 1.0.1 (a), (b)	Same as 1.0.1	Same as 1.0.1
			1.0.3 Discontinuous separation of ablative material within cell	Same as 1.0.1(a), (b)	In-plane tension	Same as 1.0.1(a), (b)	Same as 1.0.1(a), (b)	Same as 1.0.1	Same as 1.0.1
			1.0.4 Continuous bond separation along cell perimeter	Same as 1.0.1(a), (b)	In-plane tension compression, and cyclic loading	Same as 1.0.1(a), (b)	Same as 1.0.1(a), (b)	Same as 1.0.1	Same as 1.0.1
			1.0.5 Continuous separation encompassing several cells	Same as 1.0.1(a), (b)	In-plane tension	(a) Same as 1.0.1(a) (b) General area heat penetration, intensity depending on time in entry phase	Same as 1.0.1(a), (b)	Same as 1.0.1	Same as 1.0.1
			1.0.6 Ablator spalling	Same as 1.0.1(a), (b)	In-plane compression	(a) Same as 1.0.1(a) (b) Excessive ablation rate; loss of thermal protection, rate and degree depending on number of cells and time in entry phase	Same as 1.0.1(a), (b)	Same as 1.0.1	Same as 1.0.1
			1.0.7 Local damage due to outside sources with loss of ablator segments or types 1.0.1 through 1.0.6	(a) Earth orbit (b) Trans-lunar lunar orbit	Collision with LET at separation (toroidal zone Collision, during LEM/CM transposition or rendezvous docking (torroidal zone)	Depending on extent of damage, local to comprehensive loss of thermal protection capability	Minor to critical	None to catastrophic	(a) None, external repair or abort (b) Same as 1.0.1

Table 3-21. Failure Effects Analysis (Cont)

Item	Drawing Number	Function	Failure Type	Mission Phase	Cause	Failure Effect			Possible Crew Action
						On Heat Shield Performance	On Subsystem Performance	On Apollo	
2. 2 Hard lines and coaxial cable connectors	LA-4590	To provide entry points into C/M for survival liquids and gases from S/M storage areas	2. 2. 1 Tearing, fracture, and chipping of ablator about connection points.	Transearth at earth atmospheric entry	Improper removal of lines or cables during detachment of S/M from C/M	Local heat penetration with progressive loss of ablator material	Minor to critical overheating	Minor to catastrophic	None
			2. 2. 2 Separation or separations in ablator	Transearth, at earth atmospheric entry.	Excessive tension or compression of ablator due to severe two-dimensional thermal stress and entry effects	Local heat penetration	Minor to critical overheating	Minor to catastrophic	None
			2. 2. 3 Corrosion or chemical reaction of line gases and/or liquids with ablator material	Lift off, boost, earth orbit, translunar, lunar orbit, transearth	Leakage at connection areas	Depending on what and how much leakage takes place, the damage can range from local overheating to severe loss of ablator material and excessive overheating during entry conditions.	Minor to critical	Minor to catastrophic	None
2. 3 Electrical umbilical connector	LA-4583 LA-4590	To provide physical connect-disconnect point for electrical circuit between C/M and S/M	2. 3. 1 Ablator separation between pins	Transearth at earth atmospheric entry	Improper removal of connector; shear loading during removal; corroded pins	Local heat penetration	Minor to critical	Minor to catastrophic	None
			2. 3. 2 Bond separation between ablator and connector framework	All	Normal tension; separation loads	Serious heat penetration	Minor to critical	Minor to catastrophic	None
			2. 3. 3 Tearing, fracture, and chipping of ablator in or about connector	Transearth at earth atmospheric entry	Male connector is allowed to swing freely and collide with ablator material	Local heat penetration	Minor to critical	Minor to catastrophic	None

Table 3-21. Failure Effects Analysis (Cont)

Item	Drawing Number	Function	Failure Type	Mission Phase	Cause	Failure Effect			Possible Crew Action
						On Heat Shield Performance	On Subsystem Performance	On Apollo	
				(c) Trans-earth	Collision with S/M at separation				(c) None
				(d) Earth orbit, trans-lunar, lunar orbit, trans-earth	Meteoroid impact (toroidal zone)				(d) Same as 1.0.7(b)
		Thermal protection; HC provides structural rigidity to ablator	1.0.8 Honeycomb to substructure bond fails	Same as 1.0.1(a), (b)	Normal tension	Minor if area is very limited, but ablator sections may fall away if unbonded areas are large or along panel edges. Loss of thermal protection	(a) None to severe (b) None to critical	(a) None to severe (b) None to catastrophic	Same as 1.0.1
Aft Compartment 2.0 Nonhomogeneous areas 2.1 Combined shear compression pad	LA-4583	Provide thermal protection for C/M and S/M attachment point	2.1.1 Bearing failure of ablator pad at S/M loading pad	Lift off, boost	Excessive or improper preloading	Loss of shear load resistance capability S/M and C/M. Reduced ablator thickness, resulting in local overheating; this failure cannot be tolerated.	Critical, also loss of local thermal protection during earth atmospheric entry	Catastrophic	Abort
			2.1.2 Ablator crushing at substructure interface	Lift off, boost	Normal compression	Loss of shear load resistance capability between S/M and C/M. Local overheating; this failure cannot be tolerated.	Same as 2.1.1	Catastrophic, ablator crushing at compression-loaded areas (no actual pad in these three areas) with possible tipping of S/M	Abort
			2.1.3 Separation of pad material from fiberglass ring	Lift off, boost	Excessive preload failure or cyclic loading.	Loss of shear load resistance capability; this failure cannot be tolerated.	Critical	Catastrophic	Abort
			2.1.4 Fiberglass locking ring separates or fractures	Lift off, boost	Excessive preload, cyclic loading, excessive shear loads	Loss of shear load resistance capability; this failure cannot be tolerated.	Critical	Castastrophic	Abort

Table 3-21. Failure Effects Analysis (Cont)

Item	Drawing Number	Function	Failure Type	Mission Phase	Cause	Failure Effect			Possible Crew Action
						On Heat Shield Performance	On Subsystem Performance	On Apollo	
Aft Compartment 2. 4 Ablator plug and ablator plug housing	LA-4592 LA-4600	To provide thermal protection and access to aft compartment bolts	2. 4. 1 Fracture or separation of housing or plug	Lift off, boost earth, atmospheric entry	Excessive preload	Local heat penetration	Minor to critical	Minor to catastrophic	None
2. 5 Close-out members	LA-4374 LA-4377 LA-4379 LA-4381 LA-4386 LA-4574 LA-4577 LA-4579 LA-4582 LA-4584	To provide a smooth edge to HC filled ablator free edges	2. 5. 1 Separation of HC from outstanding leg	All	Loss of bond; thermal loading	Gap may result between HC and closure members, which may cause local heat penetration during reentry	Minor to critical		None
			2. 5. 2 Separation of HC from substructure leg	All	Loss of bond; thermal loading	Complete separation of ablator from substructure may occur	Minor to critical		
			2. 5. 3 Separation of gusset from closure member	All	Local expansion of ablator with respect to substructure at a free edge	As long as substructure bond holds, no over-all failure of ablator occurs			
			2. 5. 4 Separation of substructure leg from substructure	All	Thermal loading; loss of bond	Complete separation of ablator from substructure may occur			
2. 6 Joint filler (gasket)	LA-4386 LA-4583	Thermal protection of compartment and panel interface gaps	2. 6. 1 Permanent deformation	All	Mechanical and thermal loading	Local overheating; erosion of ablator			
			2. 6. 2 Erosion of ablator at interface edge	Earth atmospheric entry	Difference in ablator performances, i. e., filler material and RAD 5026-39 ablate at different rates	Local overheating; char spalling			

Table 3-21. Failure Effects Analysis (Cont)

Item	Drawing Number	Function	Failure Type	Mission Phase	Cause	Failure Effect			Possible Crew Action
						On Heat Shield Performance	On Subsystem Performance	On Apollo	
Crew Compartment 3.0 Nonhomogeneous areas—rocket nozzle extensions and other areas under study and reference design not defined to date									
4.0 Main ablator	LA-4386 LA-4589	Thermal protection	Same as item 1.0	Same as item 1.0	Same as item 1.0	Same as item 1.0	Same as item 1.0	Same as item 1.0	Same as item 1.0 except possible repairs over entire exposed area
5.0 Doors, windows, and access panels	LA-4374 LA-4377 LA-4379 LA-4381 LA-4377 LA-4579 LA-4582 LA-4584								
5.1 Ablator material	Same as item 5.	Thermal protection	Same as item 1.0	Same as item 1.0	Same as item 1.0	Same as item 1.0			Same as item 4.0
5.2 Close-out members	Same as item 5	Same as item 2.5	Same as item 2.5	Same as item 2.5	Same as item 2.5	Same as item 2.5	Same as item 2.5	Same as item 2.5	Same as item 4.0
5.3 Ablator plug	LA-4573	Thermal protection of fastener areas; provide means of gaining access to internal portion of C/M	5.3.1 Fracture, cracking, and separation of ablator plug or plugs	All	Abusive handling during prelaunch checkout; thermal loading; bond degradation	Local heat penetration	Minor to critical	Minor to catastrophic	If known, none with possible repair or abort from earth orbit

Table 3-21. Failure Effects Analysis (Cont)

Item	Drawing Number	Function	Failure Type	Mission Phase	Cause	Failure Effect			Possible Crew Action
						On Heat Shield Performance	On Subsystem Performance	On Apollo	
5.4 Joint filler (gasket)									
Forward Compartment	Same as item 2.6	Same as item 2.6	Same as item 2.6	Same as item 2.6	Same as item 2.6	Same as item 2.6	Same as item 2.6	Same as item 2.6	Same as item 2.6
6.0 Main ablator	LA-4380	Thermal protection	Same as item 1.0	Same as item 1.0	Same as item 1.0	Same as item 1.0	Same as item 1.0	Same as item 1.0	Same as item 4.0
7.0 Nonhomogeneous areas									
7.1 Abort tower wells	LA-4349 LA-4376 LA-4591	Thermal protection of abort tower and C/M interface areas	7.1.1 Cracking of ablator material	All	Thermal loading; H/S collision with abort tower	Local heat penetration	Same as item 1.0	Same as item 1.0	Same as item 4.0
			7.1.2 Separation of ablator from substructure (bond failure)	All	Thermal loading, or mechanical load due to forward compartment separation during abort	This failure cannot be tolerated. Large areas of heat penetration during earth entry may occur	Severe	Critical	None or abort
7.2 Close-out members	Same as item 2.5	Same as item 2.5	Same as item 2.5	Same as item 2.5	Same as item 2.5	Same as item 2.5	Same as item 2.5	Same as item 2.5	Same as item 4.0
7.3 Other areas undefined in present reference design									

various properties of ablative materials as a basis for quantitative determination of the inherent strength and adequacy of the design configuration. A corollary effort has been initiated to investigate the possible inclusion of the results from the IBM 640 thermal analysis program.

TEST PROGRAM

Qualification Test

An input was prepared for Volume III of the General Test Plan, SID 62-109-3. The data includes a definition of the system test program, a system logic diagram, a presentation of system criticality versus hardware requirements, and a qualification test schedule with test specimen utilization.

General Test Program Status

The development test program is 22.2 percent completed. A percentage breakdown of the major test types is as follows:

Tests	Percent
Ablation	53.0
Thermal property	18.0
Mechanical property	17.2
Structural	55.0

Qualification Status Lists

A complete change is anticipated because of the redirected test effort outlined in Revision C of the Procurement Specification. The anticipated starting date for qualification testing has been rescheduled for June 1963.

Specific Development Test

Specific development tests are shown in Table 3-22.

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Table 3-22. Specific Development Tests

Test Title	Tests Performed During Fifth Quarter	Total Tests Completed		Tests Planned For Sixth Quarter
		Number	Percent	
ABLATION				
Ascent height	7	33	22.6	10
High enthalpy	37	140	86.0	27
OVERS	82	362	118.0	
Thermal diffusivity and surface chemistry	16	39	60.0	7
Turbulence and laminar pipe	24	69	92.0	4
Two dimensional	25	25	13.0	48
Convective and radiant heating				10
Simulated trajectory heating	7	7	1.0	37
Total	198	675	53.0	143
THERMAL PROPERTY				
Conductivity	20	67	18.0	29
Specific heat	14	118	27.0	31
Heat degradation	3	23	20.0	5
Optical properties	71	71	12.4	86
Solar radiation				4
Total	108	279	18.0	155
MECHANICAL PROPERTY				
Tensile	284	707	21.5	340
Compression	47	91	8.0	135
Thermal expansion	68	196	15.0	95
Bond	135	301	23.4	145
Fastener		30		
Total	534	1325	17.2	715
STRUCTURAL				
Environmental exposure	27	27	56.5	31
Ablator to substructure bond (panels)	116	116	82.0	20
Bonded joints in ablator	15	15	20.0	34
Fastener strength in ablator	42	42	70.0	12
Total	200	200	55.0	87

Test Results: The development test results have included the final selection of the ablative material, substantiating predicted data with test data, finalizing the C/M ablator thickness, substructure selection, and preliminary environmental evaluation tests.

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SUBCONTRACTOR MANAGEMENT

The subcontractor's reliability department has been providing statistical support for Apollo heat shield material selection and evaluation in the technical areas of responsibility. This support has been (1) statistical consultation, (2) development of efficient analytical techniques and procedures, and (3) definitive statistical analyses of experimental test data and associated statistical reports.

Statistical consultation in the design of experiments has been provided in the form of review and modifications of previous plans to ascertain and establish their applicability to honeycomb-type experimental materials.

Analytical techniques and procedures for test data evaluation presented in the Fourth Quarterly Reliability Status Report, SID 62-557-4, were refined, based on their application to more definitive test programs.

Data analysis and associated statistical reports have been prepared in support of the materials test program.

LAUNCH ESCAPE

SUMMARY

Reliability data from motor development test firings have been analyzed and applied to the assessment of reliability growth. In an effort to update reliability apportionments and predictions, the entire launch escape subsystem has been reviewed. This reviewing process included a formal reliability/crew design review and a statistical evaluation of present system configuration and component designs. Basic reliability criteria did not change as a result of this review, but minor changes did occur in a few apportioned component reliability values. Table 3-23 presents a comparison of the apportioned and predicted reliability values for the launch escape

Table 3-23. Launch Escape System Apportionment and Predictions

Component	Mission Success		Crew Safety	
	Apportioned	Predicted ***	Apportioned	Predicted ***
Initiator (hot wire squib)	0.999	0.999	0.999	0.999
Launch escape motor	0.998	0.993	0.998	0.993
Pitch control motor	0.999*	0.9992*	0.999	0.9992
Tower jettison motor	0.99995	0.9998	0.99995	0.9998
Tower structure	0.99999	0.999999	0.99999	0.999999
Tower separation mechanics (explosive bolts)	0.99999	0.999999	0.99999	0.999999
Total System	0.999949	0.999994	0.996899**	0.992000**

* Not required for mission success
** Assuming boost reliability of 0.95
*** Based on state-of-the-art failure rates

system components. Results of reevaluation and studies of the explosive bolt configuration are reflected in the current system logic diagrams and mathematical models presented in Figures 3-31 and 3-32. Figure 3-33 illustrates the location of the various LES components. The tower jettison development motor test data is also being evaluated. The launch escape, pitch control, and tower jettison motors are discussed separately in this section.

RELIABILITY/CREW SAFETY DESIGN REVIEW

A launch escape system reliability/crew safety design review was conducted on 21 February 1963. Results of the design review indicated three significant problems.

Significant Problems

The problems are the following:

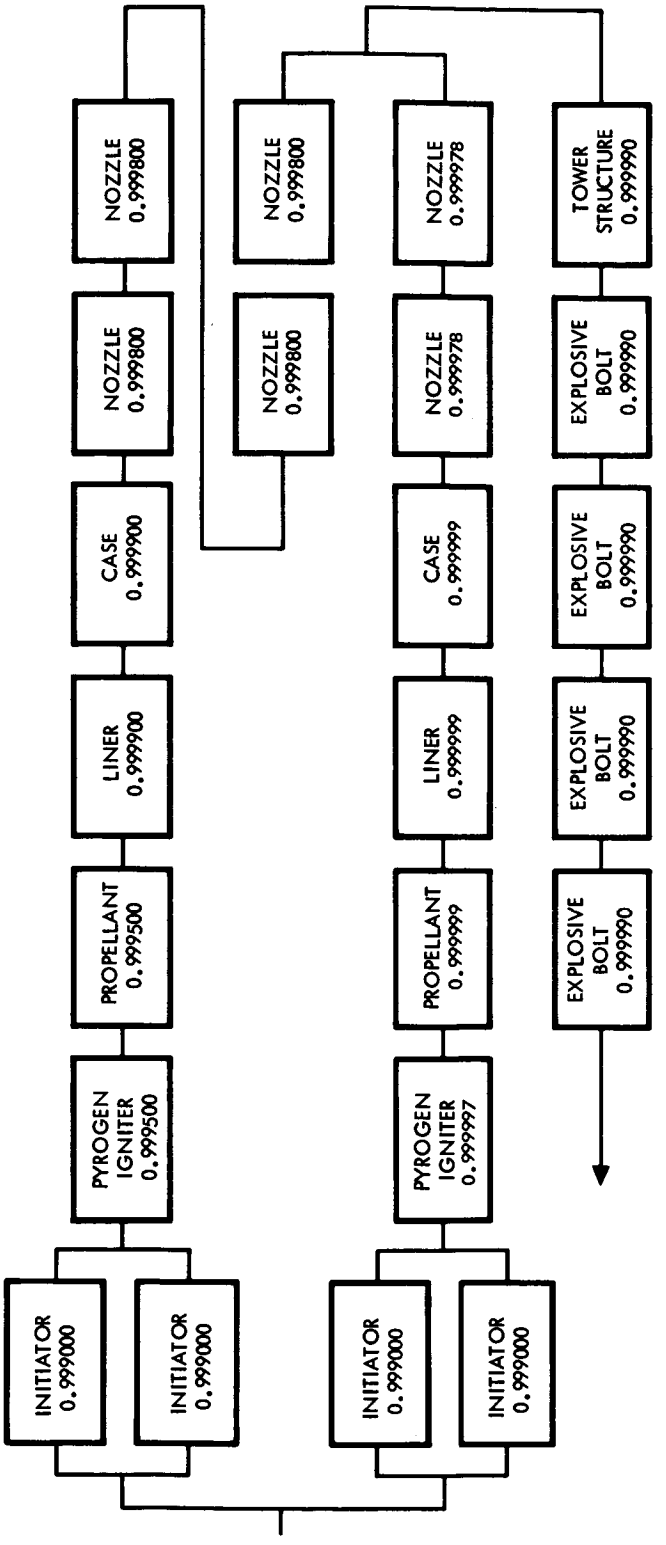
1. The apex of the launch trajectory must be determined, so that manual firing of the tower release bolts and jettison motor can be accomplished in the event of release bolt timer failure or delay. Action was assigned to the Life Sciences and the Launch Escape System Design Group.

2. There are no provisions for automatic pad abort initiation. An action assignment was made to study the possible catastrophic failure areas, the failure sensing provisions, and the use of this information in automatic abort sequencing.

3. The present design does not include a fail-safe device or sequencer that would prevent unintentional separation of the tower. Should the tower mechanical separation switch be accidentally actuated, the tower would be disconnected from the CM with catastrophic results. The necessity for fail-safe provisions is being investigated as an action item by Life Sciences and the LES Design Group.

Action Items

Table 3-24 shows the action items discussed during the design review meeting. Areas of responsibility and present status of each action item are given.



NORMAL MISSION MATH MODEL

$P_{SUCCESS} = [P_1 P_2 + P_1 (1-P_2) + P_2 (1-P_1)] P_3 P_4$

WHERE

P_1 = PROBABILITY OF SUCCESS OF LAUNCH ESCAPE MOTOR

$= P^2_{INITIATOR} + {}^2P_{INITIATOR} ({}^{1-P}_{INITIATOR})] [P_{PYROGEN\ IGNITER}] [P_{LINER}] [P_{CASE}] [P_{NOZZLE}]^4$

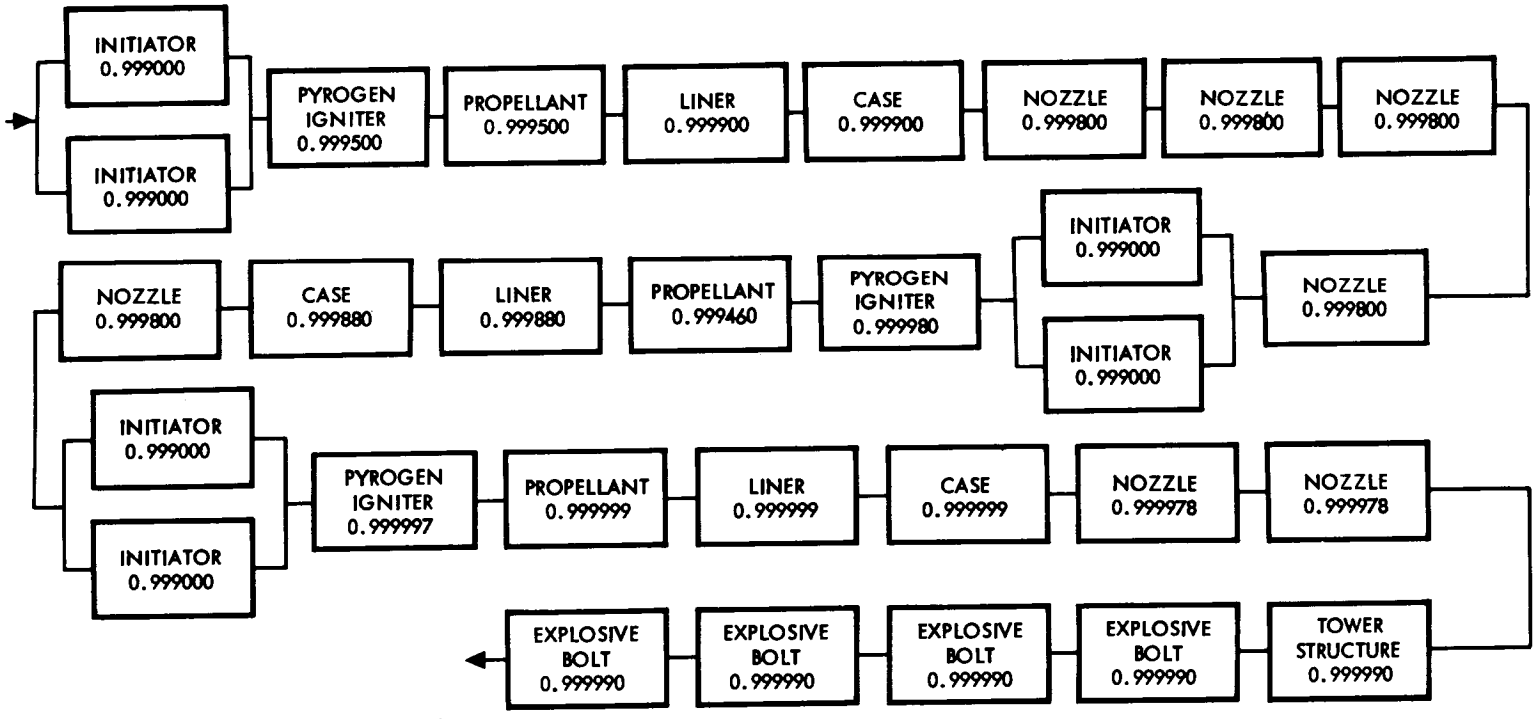
P_2 = PROBABILITY OF SUCCESS OF TOWER JETTISON MOTOR

$= [P^2_{INITIATOR} + {}^2P_{INITIATOR} ({}^{1-P}_{INITIATOR})] [P_{PYROGEN\ IGNITER}] [P_{LINER}] [P_{CASE}] [P_{NOZZLE}]^2$

$P_3 = P_{TOWER\ STRUCTURES}$

$P_4 = [P_{BOLT}]^4$

Figure 3-31. Normal Mission Tower Jettison Reliability Logic Diagram



$P_{SUCCESS} = P_1 P_2 P_3 P_4 P_5$

LAUNCH ABORT AND TOWER JETTISON MATH MODEL

WHERE

P_1 = PROBABILITY OF SUCCESS OF LAUNCH ESCAPE MOTOR

P_2 = PROBABILITY OF SUCCESS OF PITCH CONTROL MOTOR

$$= \left[P^2_{INITIATOR} + 2P_{INITIATOR}(1 - P_{INITIATOR})\right]\left[P_{PYROGEN\ IGNITER}\right]\left[P_{PROPELLANT}\right]\left[P_{LINER}\right]\left[P_{CASE}\right]\left[P_{NOZZLE}\right]$$

P_3 = PROBABILITY OF SUCCESS OF TOWER JETTISON MOTOR

$P_4 = P_{TOWER\ STRUCTURE}$

$P_5 = \left[P_{EXPLOSIVE\ BOLT}\right]^4$

Figure 3-32. Launch Abort and Tower Jettison Reliability Logic Diagram

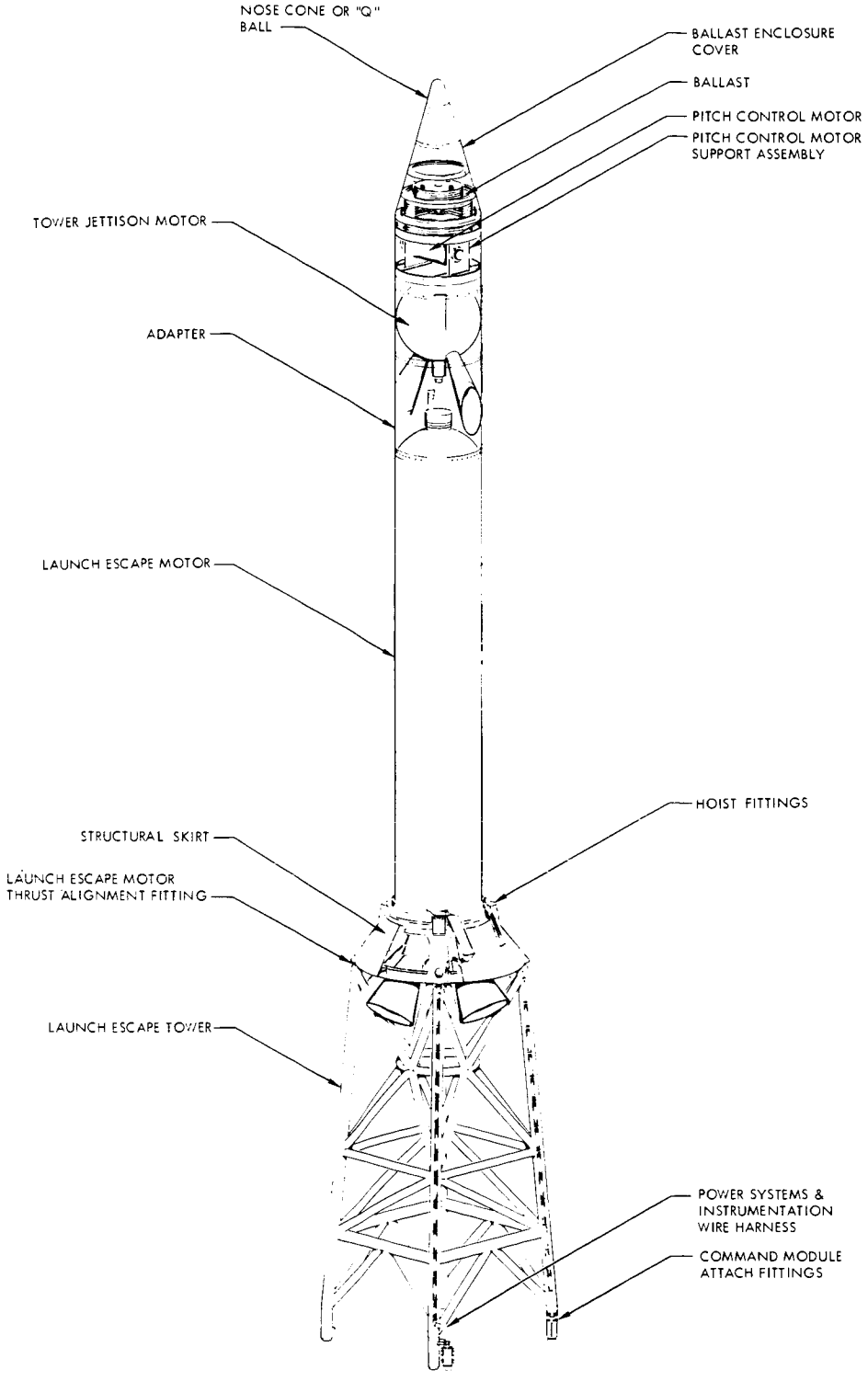


Figure 3-33. Launch Escape System

Table 3-24. Design Review Action Items

Action Item	Description of Task	Action By	Status
1. Because the hard line quick-disconnect couplings have been replaced with screw-type unions with shaped charge cutters for line severance, will the pitch control motor thrust be off-set if severance charges fail?, or should a timer be installed to delay initiation of pitch control motor until the more powerful launch escape motor has been initiated and hard line separation accomplished? If the severance charges do not fire approximately simultaneously, or fail to fire, it is likely the pitch thrust could be off-set during pad abort, especially if there is significant wind velocity or gusts.	Prepare a study on feasibility of adding a timing device.	Reliability LES design	Tests are under way to determine explosive force required.
2. Because the CM/SM electrical quick disconnect couplings have been replaced with linear-shaped charge cutters and because the CM/SM explosive separation bolts have been replaced with three tie rods, each having two shaped charge cutters, will the failure of one tie rod to separate result in a catastrophic failure?, or will a delay in separation due to partial severance of a tie rod and/or electrical cables allow the pitch motor charge to be expended before the launch tower motor completes the separation?	Prepare a study on feasibility of adding a sequencing device.	Reliability LES design	Tests for this item are included in tests for Item 1.

Table 3-24. Design Review Action Items (Cont)

Action Item	Description of Task	Action By	Status
5. At present, no provisions for an automatic abort initiation are indicated. Should areas of possible catastrophic failures be sensed and this information used for automatic abort sequence?	Areas of possible catastrophic failures to be tabulated, with their probability of occurrence, and this studied versus the time-line of sequence of events to accomplish a successful abort.	Reliability Life Science LES design	Tabulation now under way.
6. Does the ablative material on the nose cone adequately protect the parachutes from the plume of the tower jettison motor?	A study required.	Materials producibility design	No action to date.
7. It was noted that the present estimated reliability of the launch escape system motors is less than the requirements.	Study the feasibility of undertaking a reliability upgrading program on these motors.	Reliability motor design	No action to date.

Table 3-24. Design Review Action Items (Cont)

Action Item	Description of Task	Action By	Status
3. How does the crew determine the apex of the launch trajectory so that manual firing of the tower release bolts and the tower jettison motor can be initiated in event the release bolt 15.5-second timer fails or delays? The only means of determining apex of trajectory is the altimeter. Crew member must watch and then initiate action. Many other crew functions must be performed at this time, such as dump fuel, release parachutes, adjust seats, etc. If the apex of the trajectory is only 4000 feet, the elapse of seconds is critical.	Restudy the time-line of sequence of events and redesign if indicated	Life Sciences LES design	No action to date
4. Will the heat shield 3-second timer, initiating the forward heat shield explosive bolts and thruster charges, allow sufficient time for firing of the drogue mortar, pilot chute mortars, drogue release cutters, and main chute reefing cutters with sufficient altitude for the main chute to fill and brake to a normal descent rate from a pad abort? The altitude attained at the apex of the trajectory is the critical factor. If the apex of the trajectory could be less than 4000 feet, then this could present a critical problem area.	The probability of having a trajectory apex of less than 4000 feet to be investigated and redesigned if indicated.	Reliability LES design	Complete redesign is now under way base based on investigation.

TEST PROGRAM

Qualification Test

Data was submitted for inclusion in Volume III (Ground Qualification Tests) of SID 62-109. The data included a definition of the systems test program, a system logic diagram, a presentation of system criticality versus hardware requirements, and a qualification test schedule with test specimen utilization. Qualification tests will commence for each of the motors in September 1963.

Development Tests

The launch escape system development test phase is approximately 35-percent complete. Six launch escape motors, six pitch control motors, and twelve tower jettison motors have been fired. Various firings have been conducted after the motor was subjected to temperature cycling, accelerated aging, and drop testing. The static tests have been conducted during ambient, altitude, and temperature exposure. The tower jettison motor development test plan has been revised because of the deletion of humidity, shock, acceleration, and random vibration test requirements.

Thiokol is currently conducting a structural stress analysis of environmental effects on performance parameters to determine the existing stress-strain relationships and classifying the results. Development tests will continue on all three motors and the tower jettison motor program is scheduled to be concluded next quarter. A major milestone to be completed in the launch escape motor program is the tower tie-down test.

Problem Areas

During the twelfth tower jettison motor development firing, a failure occurred because improper shearing of the aluminum nozzle closure caused the carbon throat insert to be torn away. Further analysis of the failure is under way, and definitive corrective action is pending until results are available.

LAUNCH ESCAPE MOTOR

Analysis

The development motor and test conditions are being evaluated to determine if the test objectives will be attained and if the development motor configuration is acceptable for reliability evaluation purposes. No changes in test criteria or conditions are anticipated.

Table 3-25 presents a comparison of predicted and apportioned reliability levels for the launch escape motor. A reevaluation of the current motor configuration has produced more accurate statistical values than were previously available.

Table 3-25. Launch Escape Motor Apportioned and Predicted Failure Rates

Component	Failure Rates/10^6 Motors	
	Apportioned	Predicted**
Initiator (hot wire squib) (2)*	1000*	1000*
Pyrogen igniter	500	2000
Propellant	500	1500
Liner	100	950
Case	100	950
Nozzles (4)	800	1600
Total Motor	2000	7000

* Not included in total motor
** Based on state-of-the-art failure rates

Special Studies

Reliability analyses have been initiated to investigate the effect of ignition delay time on system performance. The proposed design change represents a basic design adjustment and will not require changing or expanding the test program. If this change does not reduce the rise time to a more satisfactory level, Reliability Engineering will participate in further review of igniter performance versus requirements.

Problem Areas

The long thrust rise time observed in motors fired presents a potential problem area. The problem appears to be compounded by the extreme variation of delay time observed in different motors. The prescribed corrective action consists of altering the design to increase and direct the caloric output of the igniter so that a more rapid pressure rise is provided. The ignition time will not only decrease, but also variation as a function of rise time will decrease.

Planned Activities

Delivery of units D-5 and D-6, completion of the tower tie-down test, and further evaluation and refinement of the igniter nozzle will receive primary attention during the next reporting period. Also, a launch escape motor will be used on the BP 6 pad abort test.

PITCH CONTROL MOTOR

Analysis

Reliability studies have been started on the effect of ignition delay time. Analyses are like those exercised on the launch escape motor, and they will be continued as required. Table 3-26 presents a comparison of predicted and apportioned reliability levels for the pitch control motor.

Table 3-26. Pitch Control Motor Apportioned and Predicted Failure Rates

Component	Failure Rates/10^6 Motors	
	Apportioned	Predicted**
Initiator (hot wire squib) (2) *	1000*	1000*
Pyrogen igniter	20	30
Propellant	540	460
Liner	120	80
Case	120	80
Nozzles (4)	200	150
Total Motor	1000	800

* Not included in total motor
** Based on state-of-the-art failure rates

Problem Areas

Development test data revealed that thrust rise time is outside the specified limits. This condition is similar to that encountered in the launch escape motor program, and corrective action will be accomplished by increasing the caloric output of the igniter.

Planned Activities

The firing of an intermediate impulse motor and the shipment of D-4 and D-5 delivery units are scheduled. Planning necessary for the acceleration and vibration test will be started. A pitch control motor will be used for the BP 6 pad abort test.

TOWER JETTISON MOTOR

Analysis

Thirteen development motors have been fired, but data from four motors have been excluded from use in assessing reliability. Based on data from the remaining nine motors, the present assessed motor reliability is 0.85 at the 50-percent confidence level and 0.66 at the 90-percent confidence level.

Test motor AD-6 failed after 0.4 seconds of operation at 140 F. After this failure was carefully analyzed, the required engineering changes were incorporated to prevent recurrence. No changes have been made to the failure mode and effects analysis for the tower jettison motor. The AD-6 failure analysis, however, will necessitate revision of the failure modes.

Original reliability predictions and apportionments for motor components have not been affected by the failure of motor AD-6. The redesign of the nozzle closure and insert will upgrade the reliability of each. The reliability goals will be maintained at 0.99997 for the insert and 0.999998 for the closure.

Table 3-27 presents a comparison of predicted and apportioned reliability levels for the tower jettison motor. Inconsistencies between these values and those presented prior to this report are because of reevaluation and studies of the current motor configuration.

Planned Activities

A nozzle closure redesign and checkout program will be accomplished. Nozzle assembly rework will start on all developmental motors after acceptable redesign is established. The BP-6 motor will be reworked and returned to WSMR for the pad abort test.

[redacted]

Table 3-27. Tower Jettison Motor Apportioned and Predicted Failure Rates

Component	Failure Rates/10^6 Motors	
	Apportioned	Predicted**
Initiator (hot wire squib) (2)*	1000*	1000*
Pyrogen igniter	3	25
Propellant	1	10
Liner	1	10
Case	1	10
Fixed nozzles (2)	44	145
Total Motor	50	200

* Not included in total motor
** Based on state-of-the-art failure rates

[redacted]

[redacted]

EARTH LANDING

SUMMARY

Further evaluation of the single-drogue configuration shows that if the one drogue fails, a successful deployment of two of the three main chutes will constitute a successful system operation. This assumption is based on the results of the Northrop-Ventura drop tests at El Centro. These tests indicate that the deployment of two of the three main chutes will land the command module safely. Individual parachute reliability is lower, however, due to higher dynamic pressure. (Figure 3-34 shows a simplified logic diagram of the earth landing system (ELS) and Table 3-28 shows the predicted reliability figures for this concept.)

ANALYSIS

An ELS analysis indicates that if the probability of successful deployment of the individual main parachutes at high dynamic pressure is as high as 0.990, the parachute recovery portion of the ELS will successfully meet its reliability apportionment of 0.999960.

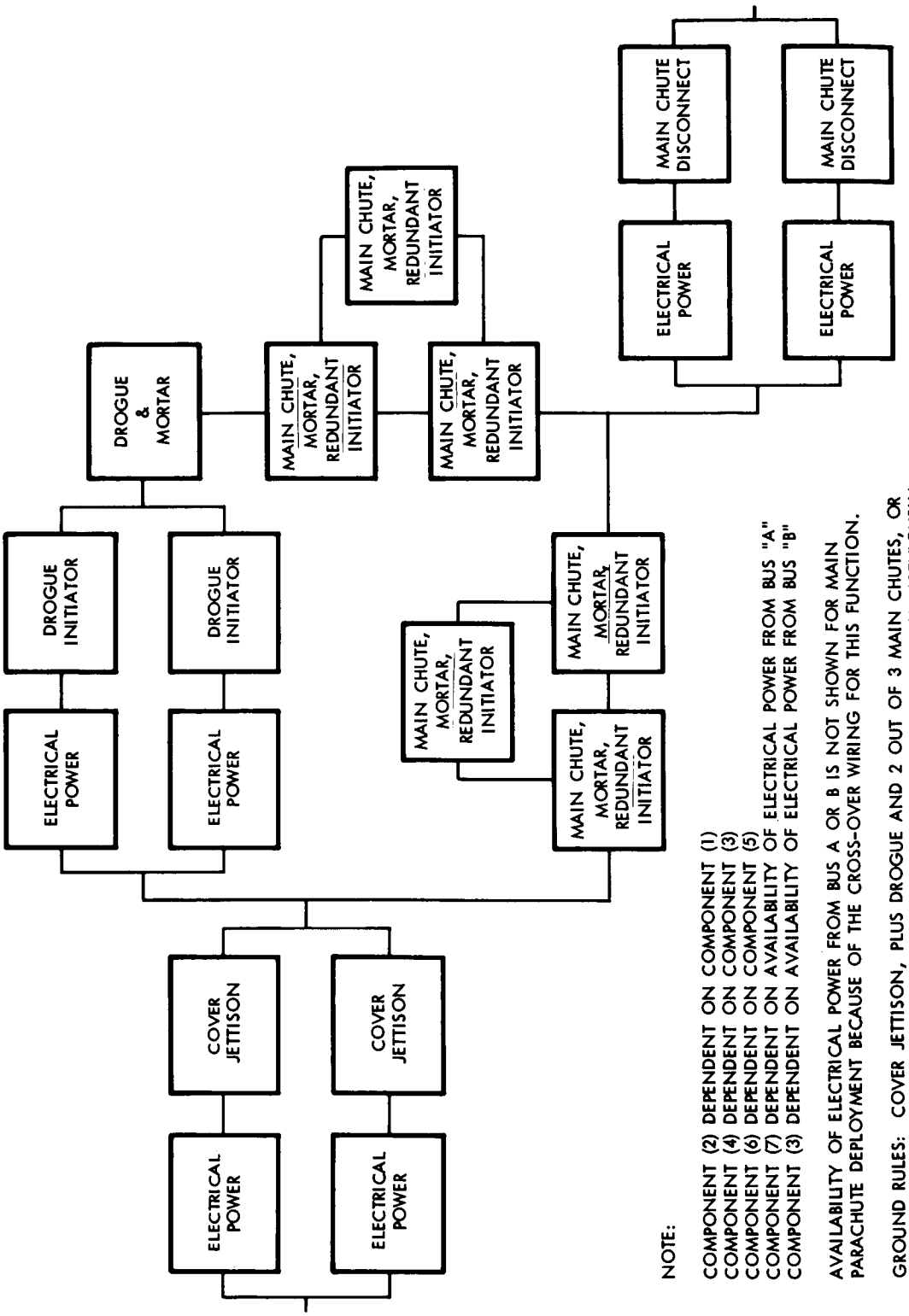
RELIABILITY/CREW SAFETY DESIGN REVIEW

A reliability/crew safety design review was held on 11 February 1963. Significant problem areas that were discussed included the following:

1. Heat shield installation. This installation, which is done after the spacecraft is positioned on the booster, requires that explosive devices be installed, adjusted, armed, and inspected in the restricted working areas at the top of the gantry. Lack of a means of adequate inspection appears to be a problem.

2. Simultaneous disreefing of the main chutes. It is possible for one chute to be deployed prior to the next one because of timing tolerances. This could damage or destroy the first chute.

3. Deletion of landing impact attenuator provisions. Deletion of provisions necessitates bracketing, supports, and other means of equipment retention that may require strengthening to ensure that items will not be torn loose upon earth impact.



NOTE:

COMPONENT (2) DEPENDENT ON COMPONENT (1)
COMPONENT (4) DEPENDENT ON COMPONENT (3)
COMPONENT (6) DEPENDENT ON COMPONENT (5)
COMPONENT (7) DEPENDENT ON AVAILABILITY OF ELECTRICAL POWER FROM BUS "A"
COMPONENT (3) DEPENDENT ON AVAILABILITY OF ELECTRICAL POWER FROM BUS "B"

AVAILABILITY OF ELECTRICAL POWER FROM BUS A OR B IS NOT SHOWN FOR MAIN PARACHUTE DEPLOYMENT BECAUSE OF THE CROSS-OVER WIRING FOR THIS FUNCTION.

GROUND RULES: COVER JETTISON, PLUS DROGUE AND 2 OUT OF 3 MAIN CHUTES, OR 2 OUT OF 3 MAIN CHUTES AT A LOWER RELIABILITY THAN PREVIOUSLY; PLUS MAIN CHUTE DISCONNECT.

Figure 3-34. Earth Landing System Simplified Logic Diagram

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Table 3-28. Earth Landing Reliability Predictions

Component	Operating Time (T56)	Predicted Mission Success Reliability	Remarks (T74)
Availiability of electrical power (7)	1 cycle	1.0	Assumed to be 100% successful for this evaluation
Availiability of electrical power (8)	1 cycle	1.0	
Cover jettison	1 cycle	0.99897	
Drogue initiator	1 cycle	0.999	
Drogue and mortar	1 cycle	0.9979	
Main chute, mortar, redundant initiator (1), (3), (5)	1 cycle	0.997899	
*Main chute, mortar, redundant initiator (2), (4), (6)	1 cycle	0.989899	Lower reliability due to high dynamic pressure
Main chute disconnect	1 cycle	0.999	
Parachute system	1 cycle	0.99998	

*Main chute reliability is assumed to be 0.990 instead of 0.998, due to higher than design dynamic pressure.

4. Removal of the impact switch. Removal of the automatic switch that disconnects the main chutes upon earth impact necessitates the use of the manual switch. Momentary crew blackout might cause severe damage or danger to the crew as a result of spacecraft tumbling due to unreleased chutes.

Action Items

Table 3-29 shows the action items discussed during the design review meeting. Areas of responsibility and present status are given on each item.

Problem Areas

The problem of the apex forward trim position has been partially solved by adding aerodynamic strakes (fins) to the yaw plane of the command module. These strakes will eliminate the supersonic apex forward trim condition and will reduce the degree of subsonic apex forward stability. For abort modes at supersonic speeds, the command module will trim in the blunt face-forward condition only. In case of a subsonic abort, the dynamics resulting from differences in command module trim at escape tower jettison are still under study.

The problem of differences in disreefing time may be solved by the parallel development program of ring sail and solid chutes. The inherent characteristic of the solid chute is slow disreefing time, which is expected to compensate for the reefing line cutter tolerance and to allow the main chutes to become disreefed simultaneously.

TEST PROGRAM

Qualification Test

Data was supplied for inclusion in SID 62-109-3, Ground Qualification Tests. The data included a definition of the systems test program, a system logic diagram, a presentation of system criticality versus hardware requirements, and a qualification test schedule with test specimen utilization.

The system qualification tests are scheduled to start June, 1963, and will be completed by January, 1964.

Development Tests

Development test program status is as follows:

Parachute drop tests, 29 percent complete
Drogue mortar and breech, 25 percent complete
Pilot mortar, 50 percent complete
Fabric tests, 33 percent complete
Main chute disconnect, 50 percent complete
Sequence controller, 23 percent complete

Table 3-29. Earth Landing System Reliability/Crew Safety Design Review Action Items

Action Item	Description of Task	Action By	Status
1. Heat Shield Installation The heat shield is installed after the spacecraft is in place on the forward end of the booster on the launch pad. This means that the explosive bolts, the mortar, and thruster pyrotechnic devices must be installed, adjusted, armed, and inspected in the restricted working area at the top of the gantry. This presents a series of problems, the most critical being the lack of a means of adequate inspection.	The installation procedure and mode of access for inspection will be re-studied to devise ways for alleviating these problems.	Recovery Systems Design Group	Study is being continued by the design group
2. Simultaneous Disreefing of the Three Main Chutes Because the parachute design is such that two chutes are required to support the spacecraft, it is vital that the main chutes be deployed at the same time. Even though three cutters are provided for each chute,	All feasible alternate designs are to be studied versus reliability and redesign undertaken accordingly.	Reliability Recovery Systems Design Group	An MCR 97 has been issued to develop a solid chute. This effort is to be carried out concurrently with the ringsail chute.

Table 3-29. Earth Landing System Reliability/Crew Safety Design Review Action Items (Continued)

Action Item	Description of Task	Action By	Status
the tolerance problem involved versus the speed with which the present chutes are deployed makes it possible for one chute to be deployed before the other.			
3. Deletion of Landing Impact Attenuators			
The landing impact attenuators have been deleted by NASA direction. Bracketry, supports, and other means of equipment retention may require beefing up to ensure that items will not be torn loose on earth impact to ricochet within the spacecraft, creating a critical personnel hazard.	A study will be made of this problem to determine whether remedial action is required.	Recovery Systems Design Group	No action to date
4. Removal of Impact Switch			
The impact switch that automatically disconnects the main chutes when the spacecraft touches the ground has been removed by NASA direction, leaving	This problem will be studied and remedial action devised if required.	Recovery Systems Design Group	No action to date

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SPACE and INFORMATION SYSTEMS DIVISION

Table 3-29. Earth Landing System Reliability/Crew Safety Design Review Action Items (Continued)

Action Item	Description of Task	Action By	Status
only the manual switch to control this function. Disconnecting the chutes immediately after impact is necessary to prevent the spacecraft from being tumbled over the terrain.			

Nineteen drop tests were conducted from 11 January through 8 March, and are as follows:

Drop No.	Test Specimen	Results
18	2 main chutes	Post-inflation interference between chutes. Inflation lag of second chute. No significant failure.
19	Drogue-mortar fired	
20	2 main chutes	Inflation lag of one chute. No significant failure
21	2 main chutes	Satisfied test objective
22	2	
23	1	
24	2	
25	2	
26	Drogue-mortar fired	
27	3 main chutes	No test. Crew member failed to arm the events controller and drag chute failed to separate from PTV and pilot chute mortars could not fire.
28	3	Successful

Satisfactory testing of shaped charge for main cluster disconnect was followed by authorization for the vendor to manufacture pre-production models for design verification tests.

Drogue Mortar Development Tests

Locked-Shut Breech Tests. The locked-shut breech tests, using a 57034-1 squib and charges of IMR 4350 powder of 70, 85, 100, 140, 170, and 200 grains, produced data which, when evaluated, indicated that a 170-grain charge would provide the energy and the burning characteristics required for sufficient exist velocity for the parachute pack. No erosion of the test breeches was evident following the firings.

Static Firing Tests. A series of eight drogue mortar test firings were made. The orifice and metering pin sizes were varied for the firings. The sixth and seventh firings were made at -70 and +160 F, respectively. The objectives of each of the eight test firings were fulfilled, and each firing was considered successful when the parachute deployed.

Problem Areas

Baroswitch failures occurred prior to and during acceptance test sequence of the controller for installation in BP 6. Two units failed during the receiving inspection test at Northrop-Ventura because of chatter. Five units failed during low-level vibration testing. A failure analysis is under investigation.

Four fusing resistors broke loose from the terminal board of sequence controller during vibration acceptance testing for BP 6. Future units will have components potted to board.

SUBCONTRACTOR MANAGEMENT

Subcontractor Activity

The following special studies were conducted by Northrop-Ventura Reliability Assurance to determine the most reliable approach in designing the sequence controller.

1. Study I. Figure 3-35 shows that four switches in a parallel series provide the most consistently reliable approach. Unless the predominant failure mode will be the open condition, with a probability greater than 7500 in 7501 ($K = 7.5 \times 10^3$), no other choice could be made.

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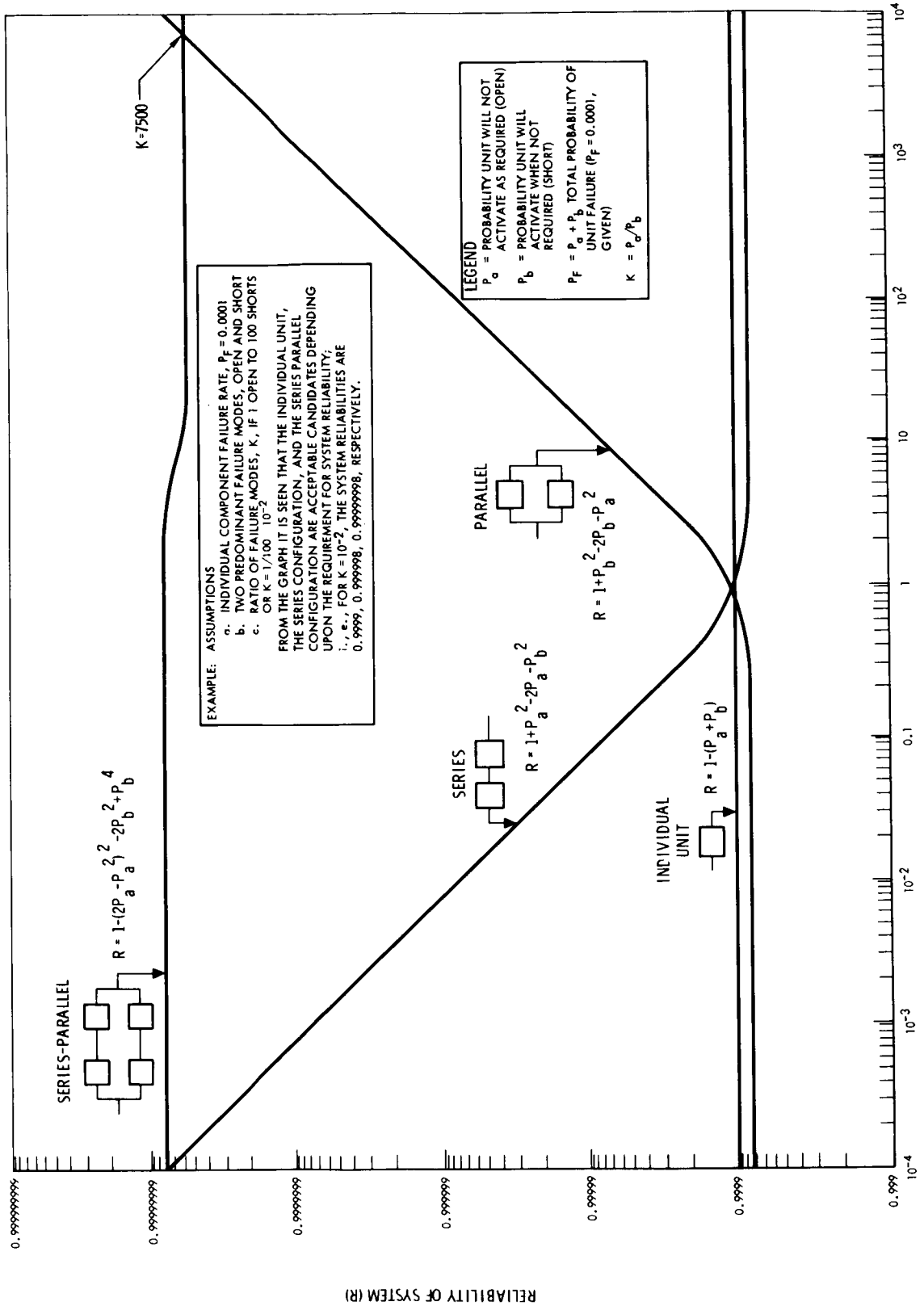
K=7500
SERIES-PARALLEL
R = 1-(2P_a - P_a^2)^2 - 2P_b^2 + P_b^4
SERIES
R = 1+P_a^2 - 2P_a - P_b^2
PARALLEL
R = 1+P_b^2 - 2P_b - P_a^2
INDIVIDUAL UNIT
R = 1-(P_a + P_b)
EXAMPLE: ASSUMPTIONS
a. INDIVIDUAL COMPONENT FAILURE RATE, P_F = 0.0001
b. TWO PREDOMINANT FAILURE MODES, OPEN AND SHORT
c. RATIO OF FAILURE MODES, K, IF 1 OPEN TO 100 SHORTS OR K = 1/100 10^-2
FROM THE GRAPH IT IS SEEN THAT THE INDIVIDUAL UNIT, THE SERIES CONFIGURATION, AND THE SERIES PARALLEL CONFIGURATION ARE ACCEPTABLE CANDIDATES DEPENDING UPON THE REQUIREMENT FOR SYSTEM RELIABILITY; i.e., FOR K = 10^-2, THE SYSTEM RELIABILITIES ARE 0.9999, 0.999998, 0.99999998, RESPECTIVELY.
LEGEND
P_a = PROBABILITY UNIT WILL NOT ACTIVATE AS REQUIRED (OPEN)
P_b = PROBABILITY UNIT WILL ACTIVATE WHEN NOT REQUIRED (SHORT)
P_F = P_a + P_b TOTAL PROBABILITY OF UNIT FAILURE (P_F = 0.0001, GIVEN)
K = P_a/P_b
RELIABILITY OF SYSTEM (R)
0.999999999
0.99999999
0.9999999
0.999999
0.99999
0.9999
0.999
10^-4
10^-3
10^-2
0.1
1
10
10^2
10^3
10^4
RATIO OF FAILURE MODES, K = P_a/P_b

Figure 3-35. Optimum Altitude Switch Configuration Reliability of System Versus Ratio of Failure Modes (Fail Open/Fail Short)

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2. Study II: Figure 3-36 shows that an insignificant network reliability gain is achieved when the individual component values are greater than 0.999. Results of this study have been used to select the most reliable circuit design for the Apollo sequence controller.

Northrop-Ventura is now working on math models for the single-drogue configuration, which include estimates of main chute reliability at a dynamic pressure of 90 to 96 q. Northrop-Ventura and NAA Engineering and Reliability have agreed that crossovers should be used in the sequence controller for main chute deployment because of the baroswitch tolerances. However, crossovers should be mechanical rather than electrical. The tolerance of the 15,000-foot main parachute baroswitch is ±365 feet, which would allow one channel of the sequence controller to be armed and would deploy the main chutes well ahead of the other channel. This time difference would be important if one chute's squib failed to function. It would then be several seconds before the other channel would become armed and deploy the chute. Simultaneity of main chute deployment is desirable and because of the tolerance of the baroswitches, crossover wiring for this function is necessary.

PLANNED ACTIVITIES

A method of determining crew survival for earth-landing conditions is being developed. The method will be based on the available statistical information on wind speed and vertical descent speed of the command module. For this purpose, equations that can be integrated directly are being derived to fit the normal curve. If the equations are successfully derived, a graph showing the probability of various horizontal and vertical command module landing speeds will be constructed. The probability of earth-landing conditions exceeding the emergency limits will then be determined from the graph after an envelope of emergency conditions has been superimposed.

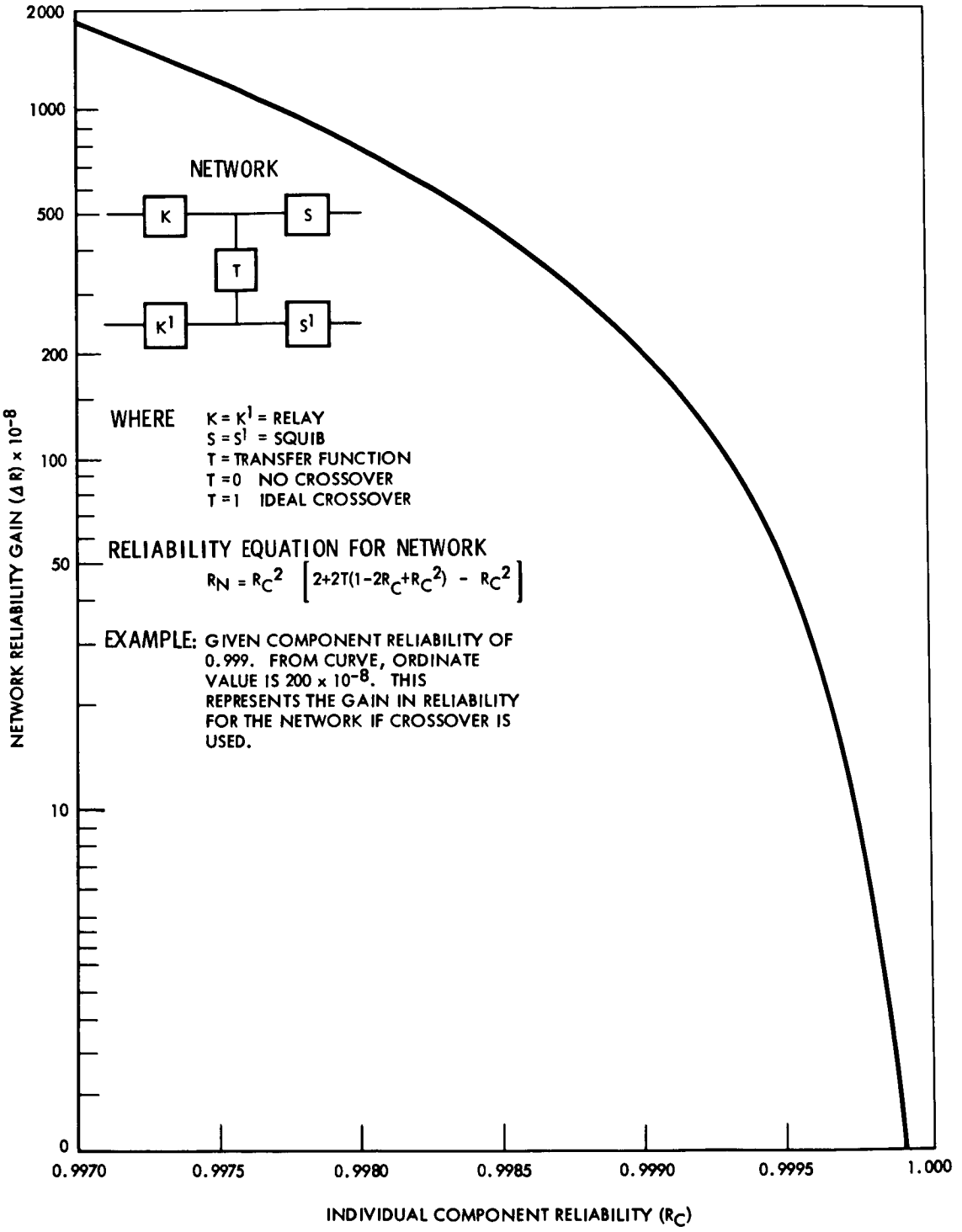


Figure 3-36. Network Reliability Gain (ΔR Network) Versus Individual Component Reliability for R_{squib} R_{relay}

OTHER STUDIES

LEM INTEGRATION

The Fourth Quarterly Reliability Status Report (SID 62-557-4) defined the LEM transposition and docking studies. This report continues with reliability evaluations of design concepts for impact attenuation. Subsequent reports will include analysis of selected systems and the components making up these systems.

Analysis

A reliability study was performed that evaluated design concepts for the command module/lunar excursion module impact attenuation system. Four proposed modes were evaluated, and the results indicate that none of the concepts were capable of meeting the apportioned failure rate of 28 per 10^6 missions. The methods evaluated were the following:

Mode 1a. Hydraulic actuator for a single-top hatch LEM

Mode 1b. Hydraulic actuator for two-hatch LEM

Mode 2a. Annular dash pot

Mode 2b. Modified annular dash pot

The predicted reliability values for mission success are shown in Table 3-30. These values were calculated using component reliabilities that refer to inherent properties – to integrity at the component level.

Table 3-30. Reliability Analysis Results

Configurations	Mission Success	
	Apportioned	Predicted
Mode 1a	0.99972	0.999865
Mode 1b	0.99972	0.999860
Mode 2a	0.99972	0.999935
Mode 2b	0.99972	0.999930

Shock Attenuation Concepts

Mode 1a. Hydraulic actuator mode (for a top hatch only LEM). The mechanism of this mode is an assembly that consists of (1) a locating cone with short cylindrical extension at inside diameter end, (2) a cylindrical section attached to structure adjacent to LEM top hatch and over which the locating cone cylindrical section slips, (3) an annular metal bellows, one end attached near the base of the locating cone, the other to upper structure of LEM, and (4) four hydraulic actuators equally spaced and inside the annular metal bellows (these effect final closure after capture). Axial impact loading is reacted by the annual bellows. Lateral impact loading is reacted by the locating cone and cylindrical sections (shear).

Mode 1b. Hydraulic actuator mode (for a two-hatch LEM). The mechanism in this mode is an assembly that consists of (1) two locating cones, each as described under Mode 1a, (2) a cylindrical interconnector over which each locating cone slides (one at either end), (3) an annular metal bellows with each edge attached near the base of the opposed locating cones, and (4) four hydraulic actuators equally spaced and inside the annular bellows, bases and piston rods attached to the opposed locating cones near their respective bases. The axial impact load is absorbed by the annular bellows. Lateral impact loading is reacted to by the locating cone and cylindrical sections (shear). Some impact energy may be dissipated by permanent deformation of the locating cone at impact. In operation, one end of the cylindrical interconnector is latched at either LEM or the CM side, the free and locating cone being automatically deployed to accept impact.

Mode 2a. Annular dash pot mode. The mechanism in this mode is an assembly that consists of (1) an annular bellows, one side attached to the cylindrical section that functions as piston rod for the donut-shaped, flat piston for the (2) annular dash pot, (3) two locating cones, one mounted on the base of the annular dash pot and the other mounted on the annular metal bellows. Items (1), (2), and (3) form an assembled unit and together with (4) a cylindrical interconnector, react to axial loads. As in Mode 1b, in operation the cylindrical interconnector is latched at one end to LEM or to CM leaving the free-end locating cone automatically deployed to accept impact.

The initial axial impact compresses the annular bellows, and these react against the annular dash pot. Final closure under RCS power (SM in translunar, LEM in lunar) compresses the partially expanded annular bellows. Lateral impact loads are reacted to by the bellows and cylindrical piston rod section (shear).

Mode 2b. Modified annular dash pot mode. The mechanism in this mode is an assembly that consists of the same elements as Mode 2a except for a modification to minimize assembly length. The modification is a cylindrical section (added) located immediately outside the annular bellows and attached to the free end of the bellows by a suitable fitting; the opposite edge is attached to the annular cylindrical piston rod by means of fittings.

Functioning of the Mode 2b mechanism is similar to Mode 2a. The logic diagram for Mode 2b is identical to that of Mode 2a except for the factor of the added cylindrical section.

Both Modes 2a and 2b complicate the problem of sealing due to an added interface.

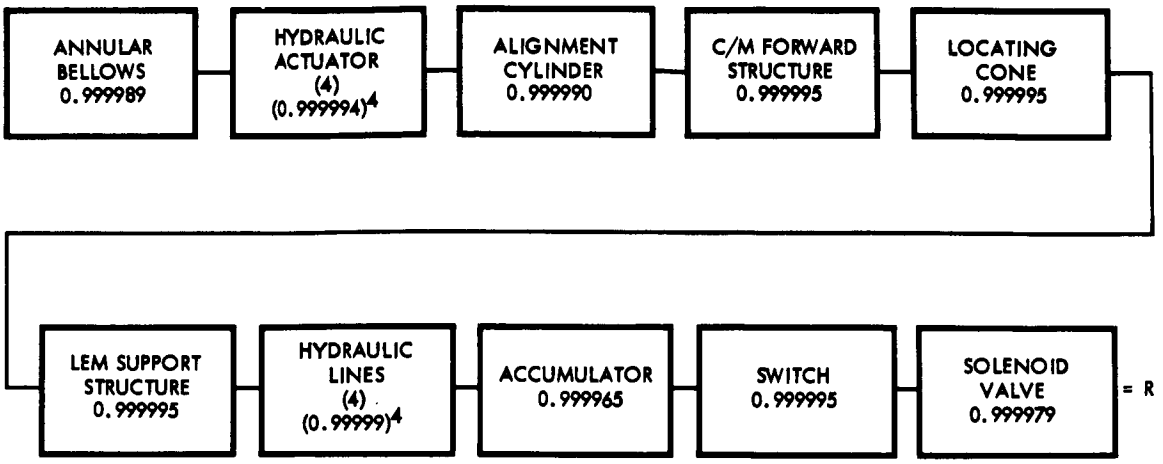
Figures 3-37, 3-38, and 3-39 show reliability logic diagrams of these modes for the impact attenuation.

Each mode of impact attenuation proposed employs a spring-type annular bellows. Modes 1a and 1b do not provide a means for eliminating energy of translation, but would redistribute it in the combined LEM-CM-SM system. Modes 2a and 2b provide for eliminating a small part of the translational energy. The remaining energy is distributed and retained in the system.

An impact attenuation system must provide means by which the relative level of translational energy between the impacting bodies is eliminated in order to reduce impact forces between two bodies to near zero. One means of dissipating translational energy is by converting it to heat by permanently deforming metal, shaving or milling metal, rubbing two surfaces together, or forcing fluid thru an orifice. Compressing a spring or bellows does not remove energy from a system but retains it as a reaction potential.

Some problem areas uncovered during this reliability analysis were:

1. Dissipation of translational energy (impact)
2. Rebound after impact
3. Effectiveness of locating cone
4. Degree of positioning accuracy required for docking - due to close tolerance mating

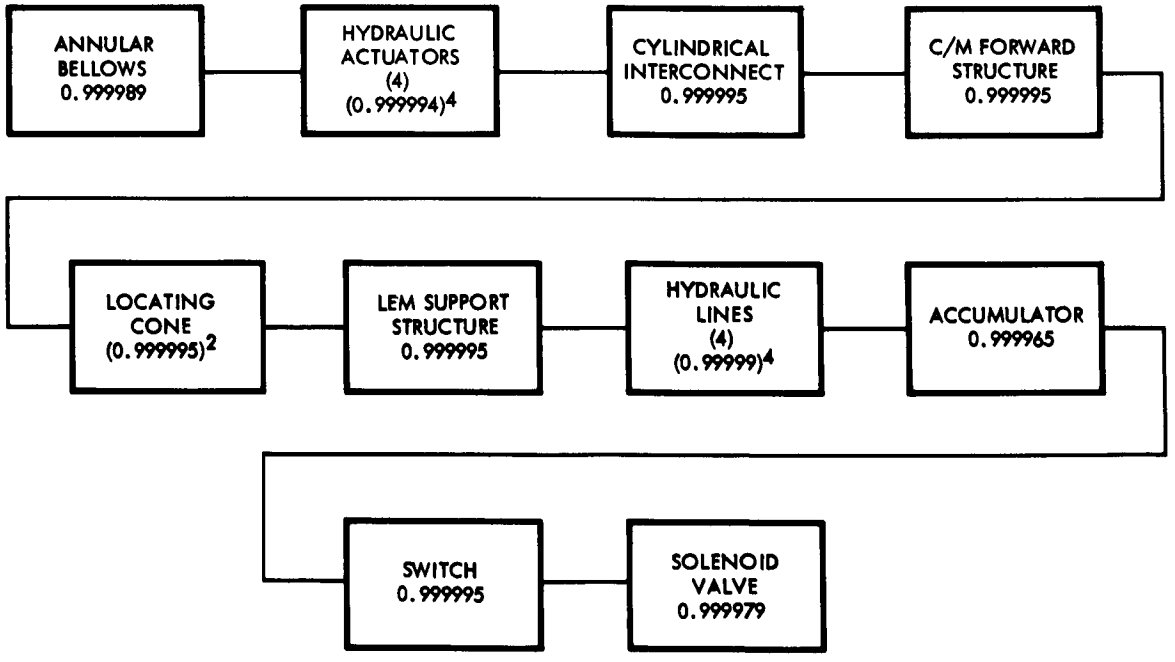


$$R = (0.999989)\ (0.999994)^4 \cdot (0.999990) \cdot (0.999995)^3 \cdot (0.999980)^4 \cdot (0.999965)$$

$$(0.999995) \cdot (0.999979) = 0.999865$$

NOTE: THE CALCULATION ABOVE ASSUMES A SINGLE IMPACT FOR DOCKING AND IGNORES POSSIBILITY OF REBOUND AFTER IMPACT.

Figure 3-37. Hydraulic Actuator (Mode 1a) Logic Diagram



$$R = (0.999989)\ (0.999994)^4 \cdot (0.999995)^5\ (0.99999)^4 \cdot (0.999965) \cdot (0.999995)\ (0.999979)$$

Figure 3-38. Hydraulic Actuator (Mode 1b) Logic Diagram

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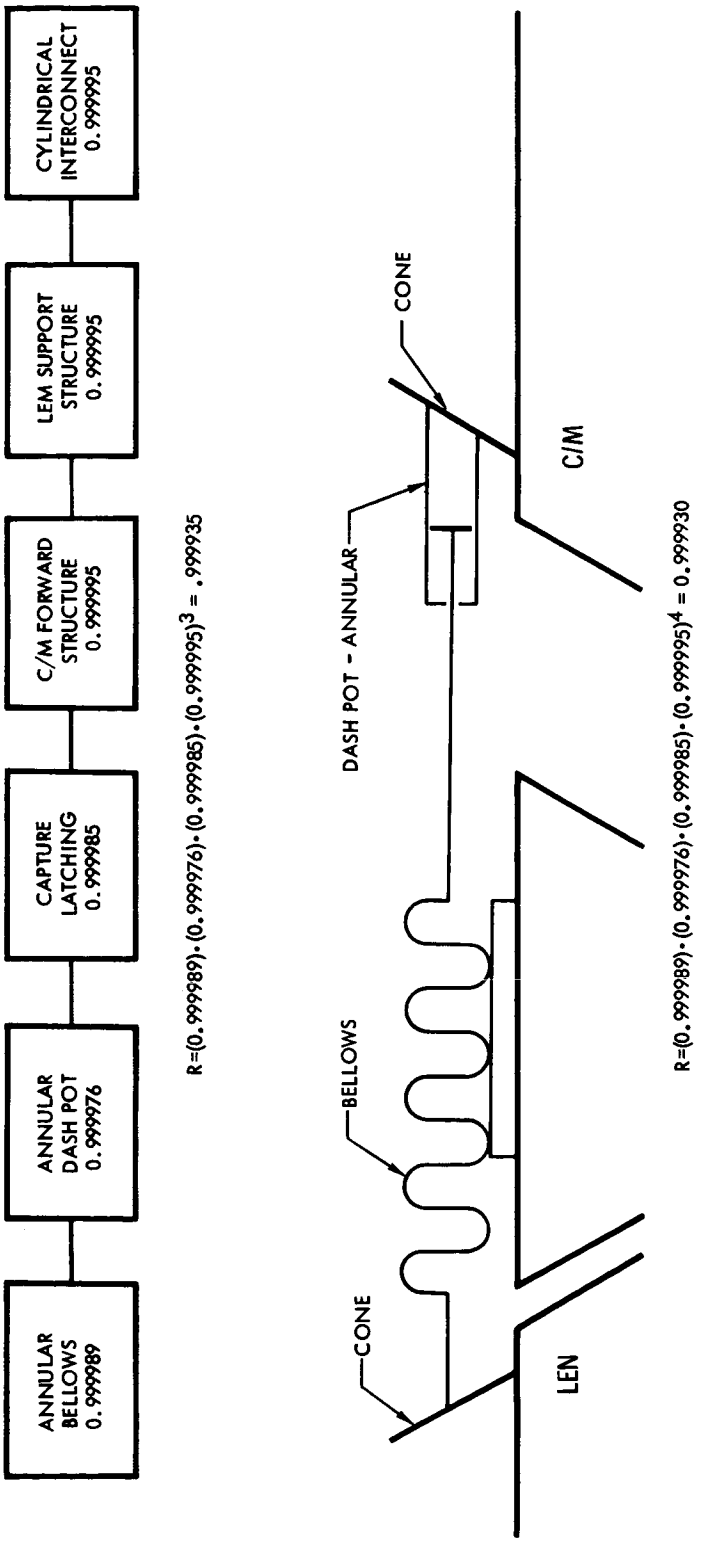


Figure 3-39. Annular Dash Pat (Modes 2a and 2b) Logic Diagram

5. Synchronizing hydraulic actuators

6. Hydraulic fluids - in space temperature environment

7. Annular metal bellows - feasibility

8. Need for automatic centering and aligning tendency during docking as an inherent functional property of the shock attenuation system and seals

SEPARATION SYSTEMS

All of the spacecraft separation systems are based on using redundant initiators to ignite a pyrotechnic device that in turn fails a structural tie. The same type of initiator is used throughout and has a predicted failure probability of 0.001. When used redundantly, as is the case in all Apollo applications, the probability of failure decreases to 0.000001.

Launch Escape Tower Separation System

The present configuration for separating the tower from the command module, Figure 3-40 consists of four explosive bolts with a removable cartridge on each bolt. Each cartridge is comprised of dual hot-wire initiators and a main charge. The hot-wire initiator has an electrical connector, bridgewire, primary explosive, and secondary explosive. The cartridge can be removed and checked away from the command module.

The predicted system reliability for successful operation of all four bolts is 0.999996, which exceeds the apportioned reliability of 0.999995.

The predicted reliability is based on the assumption that all four bolts must operate. During a normal tower jettison the launch escape motor could be utilized to break a failed bolt, but this would constitute an abort, since the ablative material surrounding the failed bolt would likely be crushed as the bolt is broken because of a combination of axial and bending loads. Since a failed bolt would constitute an abort, utilizing the launch escape motor to break a failed bolt would improve crew safety but would not affect mission success. For an abort which occurs before normal tower jettison the launch escape motor could not be utilized since the motor would have burned out before tower jettison. Reliability studies of the tower separation system are continuing in co-ordination with Design Engineering.

Forward Compartment Heat Shield Separation System

The four explosive bolts previously used in the forward heat shield separation system have been replaced with two tension failure bolts. (See

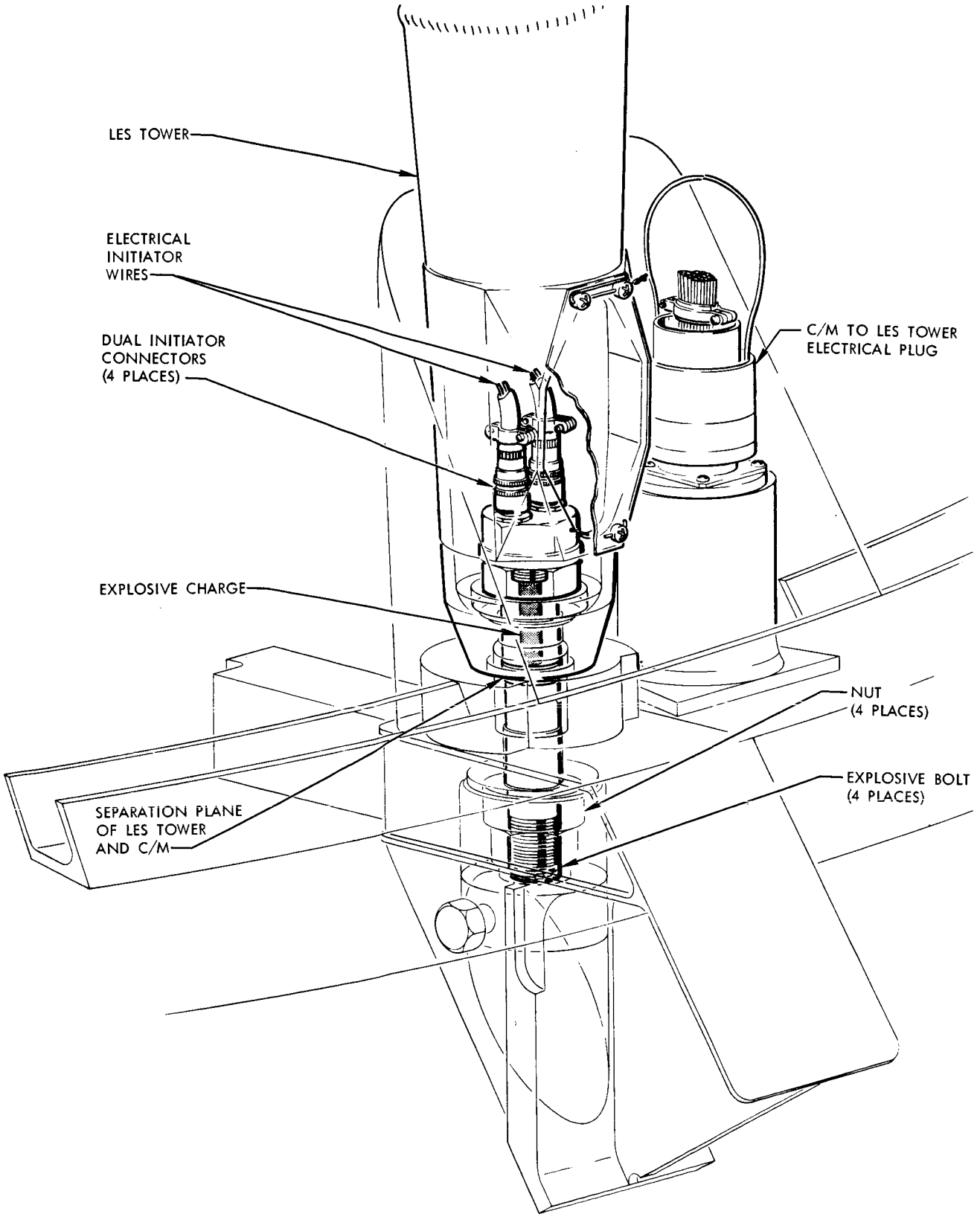
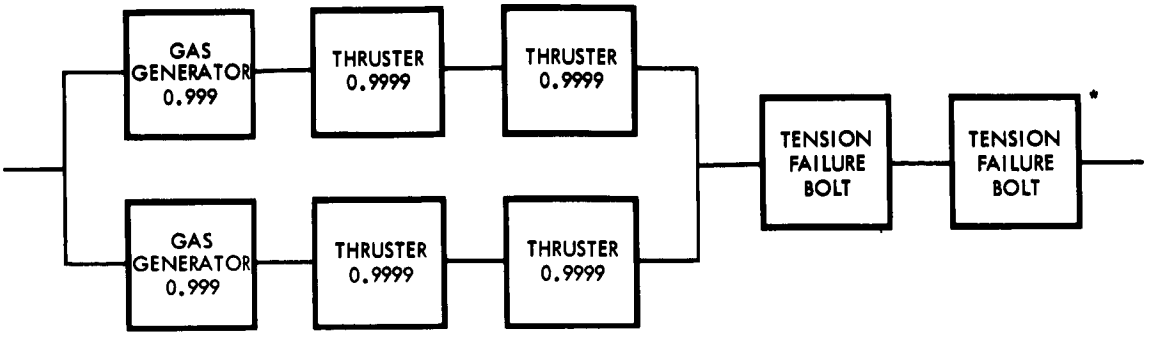

LES TOWER
ELECTRICAL INITIATOR WIRES
DUAL INITIATOR CONNECTORS (4 PLACES)
C/M TO LES TOWER ELECTRICAL PLUG
EXPLOSIVE CHARGE
NUT (4 PLACES)
EXPLOSIVE BOLT (4 PLACES)
SEPARATION PLANE OF LES TOWER AND C/M

Figure 3-40. Launch Escape Tower Separation (Bolt, Typical, Four Places)

Figure 3-41.) The predicted reliability of the system is 0.9999985, which exceeds the apportioned reliability of 0.999995. Using tension failure bolts simplifies the previous system, resulting in an increase in reliability. A logic diagram of the forward heat shield separation system is shown below.



*The tension failure bolts are not included in calculating the reliability, since neither bolt has a function within itself.

Each bolt is designed to fail at 6135 ± 5 pounds. Each of the four thrusters are designed to peak at about 9500 pounds thrust. If one of the redundant gas generators fails, the two thrusters that would be actuated will provide 19,000 pounds thrust. This is a 55 percent margin over the thrust required to break the two bolts.

Command Module to Service Module Separation System

Flexible linear shaped charges, Figure 3-42 are used to cut the three straps that attach the command module to the service module. The predicted reliability is 0.999997, which exceeds the apportioned reliability of 0.999995. Two flexible linear shaped charges are used on each strap, either of which will cut the strap. These charges are both initiated at each end by dual initiators. Reliability studies of the command and service modules separation system are being continued.

Adapter Separation Systems

A reliability evaluation was made of the system separating the service module adapter and lunar excursion module adapters. A schematic of the system is shown in Figure 3-43 and the sequence of operation appears in Figure 3-44. Each of the adapters uses flexible linear shaped charges along the adapter structure. The linear shaped charges on each adapter are detonated by dual squib-ignited mild detonating fuses. A linear-shaped charge has an initiation point at each end, but there are several ways of reaching each end due to the interconnection of the shaped charges. For this reason, the

reliability of separating each adapter is approximately equal to the reliability of the dual squibs. Since the reliability of one pair of dual squibs is 0.999999, the reliability of separating the service module and lunar excursion module adapters is 0.999997. This exceeds the apportioned reliability of 0.999995.

Studies will be continued on the adapter separation systems, and design changes or improvements will be suggested when necessary to improve reliability.

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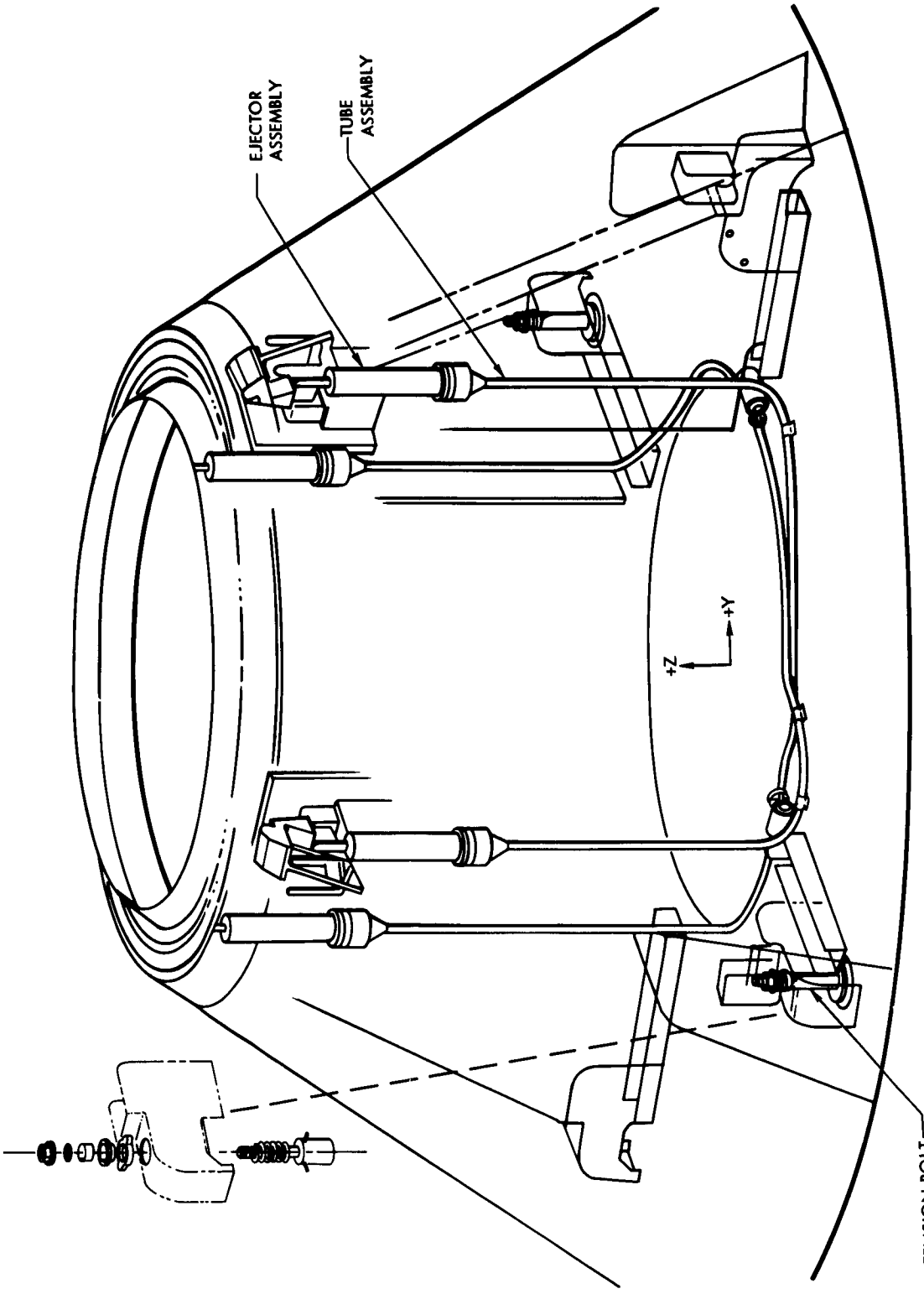

Figure 3-41. Forward Compartment Jettison

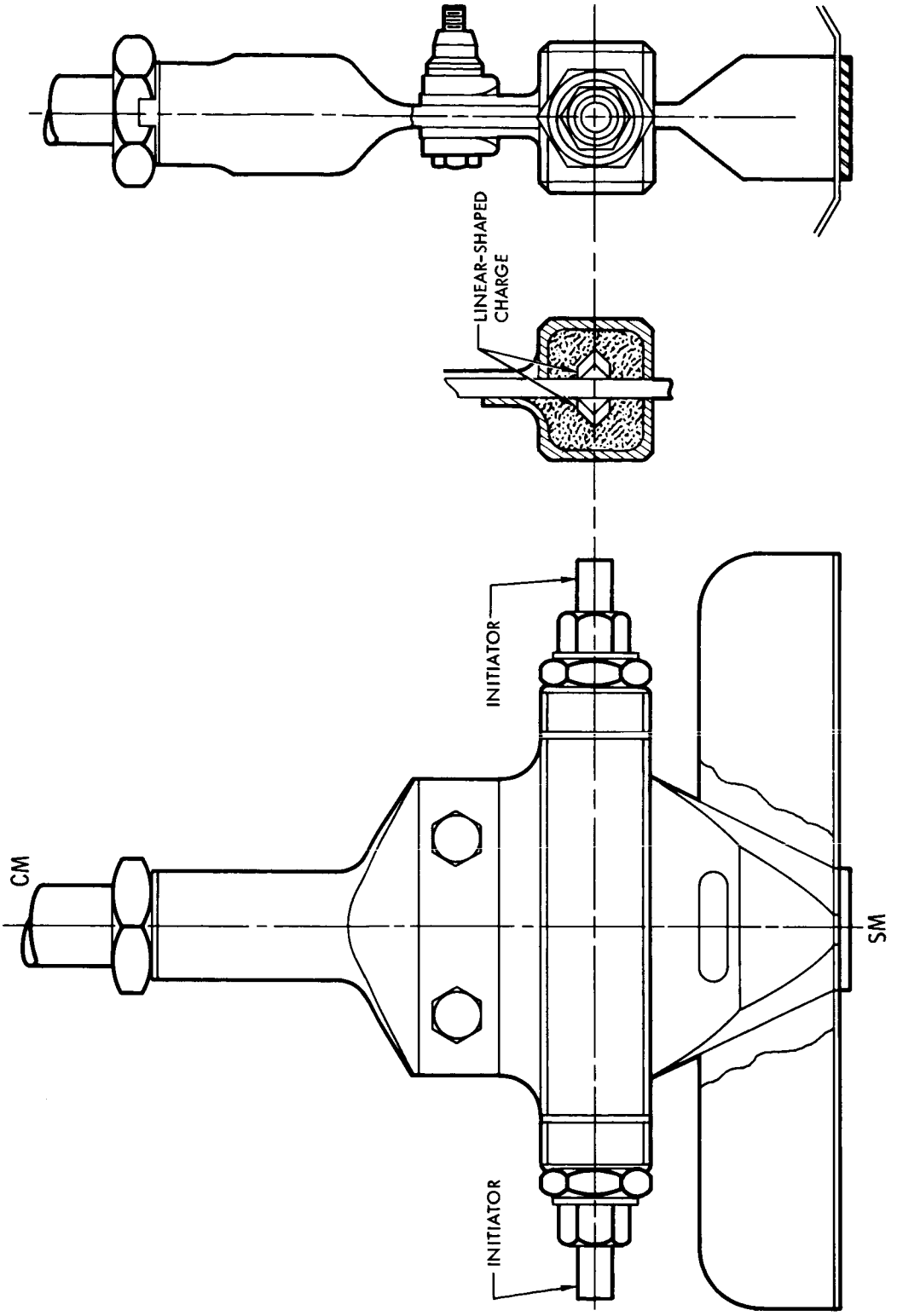

Figure 3-42. Command Module/Service Module Separation (Typical, 3 places)

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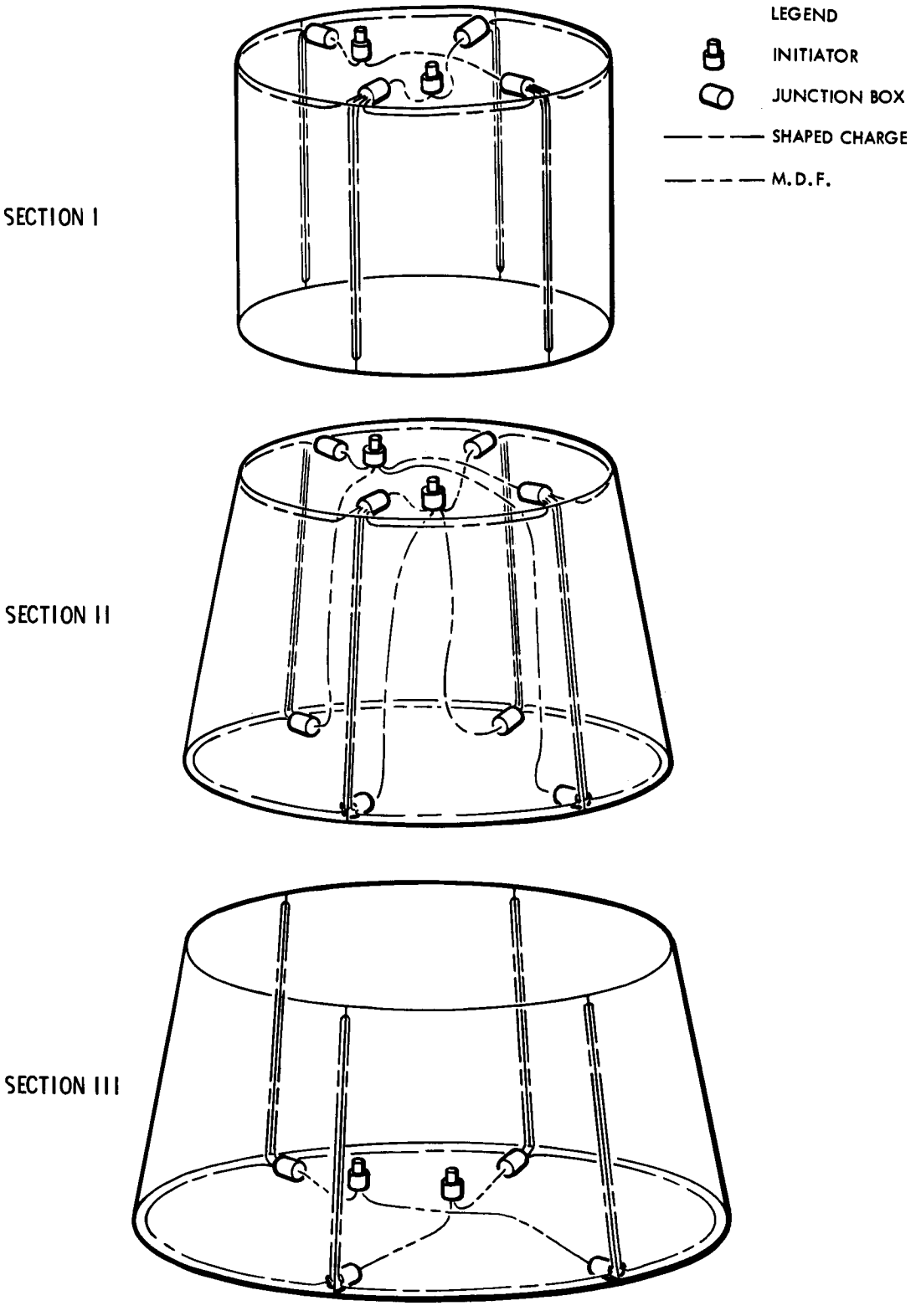

Figure 3-43. Adapter Separation System—LOR Mission

[redacted]

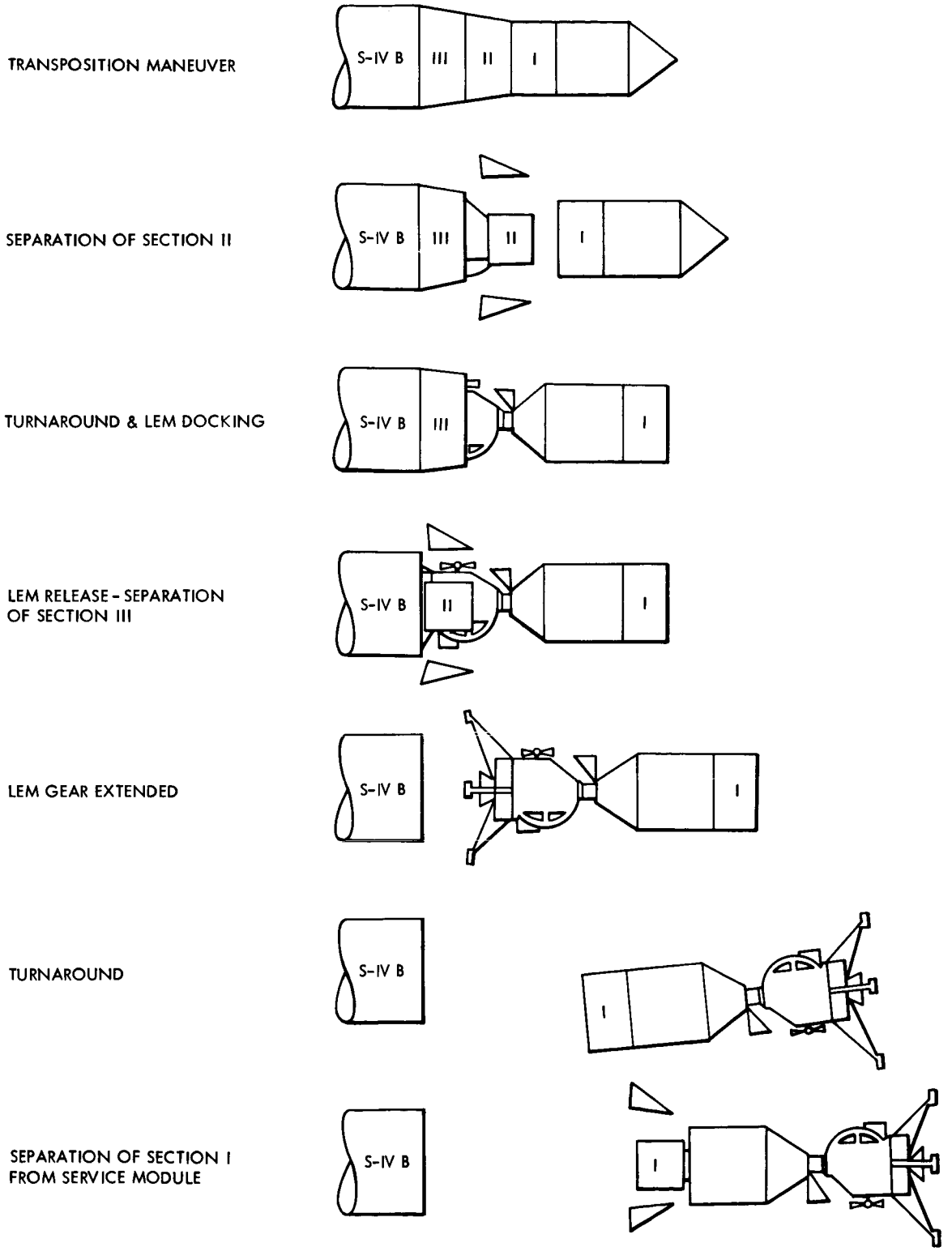

Figure 3-44. Adapter Separation Sequence

[redacted]

IV. ELECTRONIC SYSTEMS ANALYSIS

SUMMARY

Electronic systems analyses are continuing with updated and more detailed logic diagrams at lower levels of assembly. These diagrams are being expanded to include the physically oriented and the functionally sequenced relationships between the spacecraft electronic subsystems at the module level; current diagrams reflect physical and functional relationships as determined by performance requirements. Anticipated changes in requirements, as for example, the navigation and guidance and stabilization and control subsystems serving as alternate modes rather than as monitoring functions for the S-IVB guidance and control, will necessitate corresponding changes in the logic of the applicable mission phases. Such changes will be initiated as changes in requirements become formalized. Mission profile time lines have been expanded to conform to specific equipment operations. In addition to the method of probability density functions, symbolic logic is being used to check the accuracy of the reliability evaluations as determined by the Monte Carlo mission simulation machine program.

Also, evaluations are being conducted of the influence of the man-in-the-loop on the assurance of equipment availability through maintenance actions and spares replacements.

ANALYSIS

Subsystem reliability analyses are continuing based upon the expanded LOR time line and detailed logic diagrams. The spacecraft subsystems logic diagrams have been modified to maintain functional identity and compatibility with the expanded LOR mission profile and to implement evaluations at the module level. Although the Monte Carlo machine program or mission simulation technique has been selected as the basic mode of analyses, symbolic logic, combinations of successes and failures, and probability density function methods to spot check the accuracy of the mission simulation program results are being used.

Logic diagrams illustrated in Figures 4-1 through 4-18 represent Apollo electronics safe abort configurations for respective phases of the LOR mission as determined by the current mission profile time lines. Interface areas between electronics and electro-mechanical subsystems are being investigated. Extensive updating of the existing system logic is required

because of recent design changes in the in-flight test system, stabilization and control subsystem, and communications and data subsystem.

Presentations of spacecraft logic diagrams, subsystems status, sparing requirements, and machine program techniques were made to the Design Review Board by Apollo Reliability Engineering on 30 January 1963. Information is being requested from subcontractors and the associate contractor to aid in the implementation of a more detailed spares analysis at the subassembly level. These data will be presented to the design engineering groups.

Failure Effects Analysis

The failure effects analyses of spacecraft subsystems are being updated to reflect functional changes based upon the receipt of more detailed information. These analyses indicate the equipment functions, the failure modes, the effect of each failure mode on the subsystem and spacecraft, and the possible crew action in the event of failure. These analyses are shown in the Instrumentation, Communications and Data, Stabilization and Control, and Guidance and Navigation Subsystems sections in this report.

This section describes the expansion of the spacecraft electronics subsystems from the equipment to the module level and the corresponding changes in analytical techniques necessary for reliability evaluation purposes. Module weights and failure rates are used to arrive at an index of sparing required to assure the attainment of the required probability values when incorporated into the logic diagrams for mission success and safe abort.

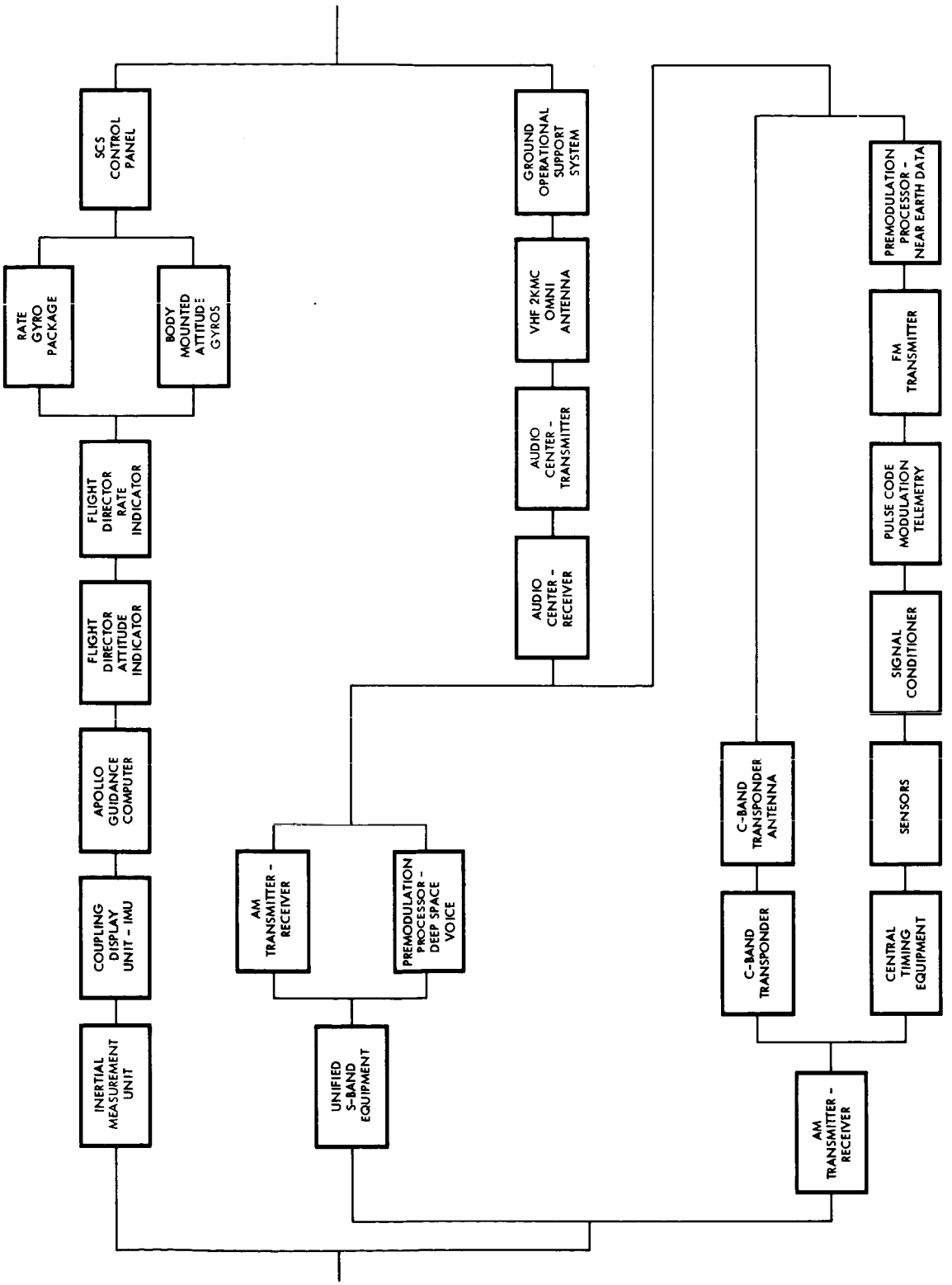

Figure 4-1. Launch to Earth Orbit

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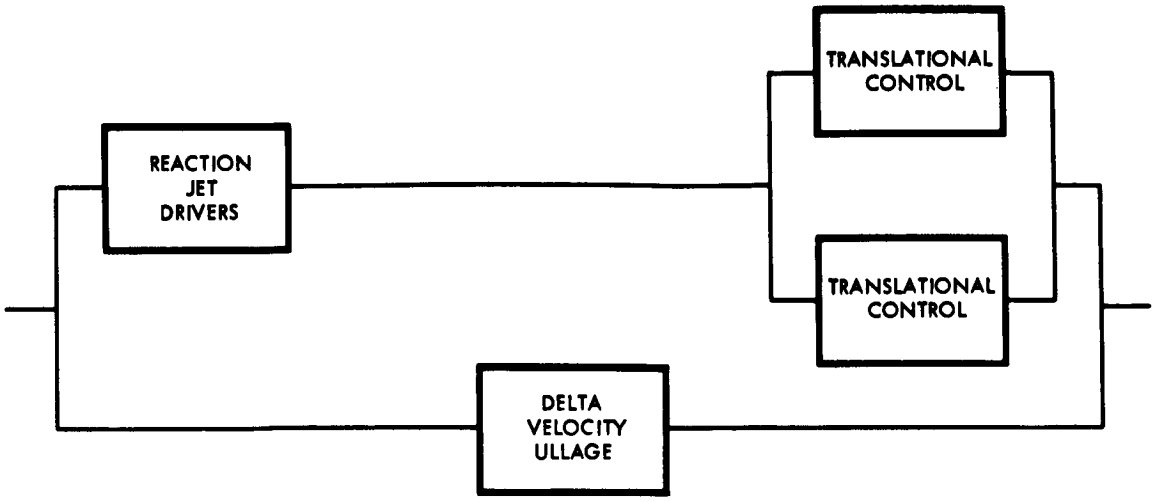

Figure 4-2. S-IV B Separation

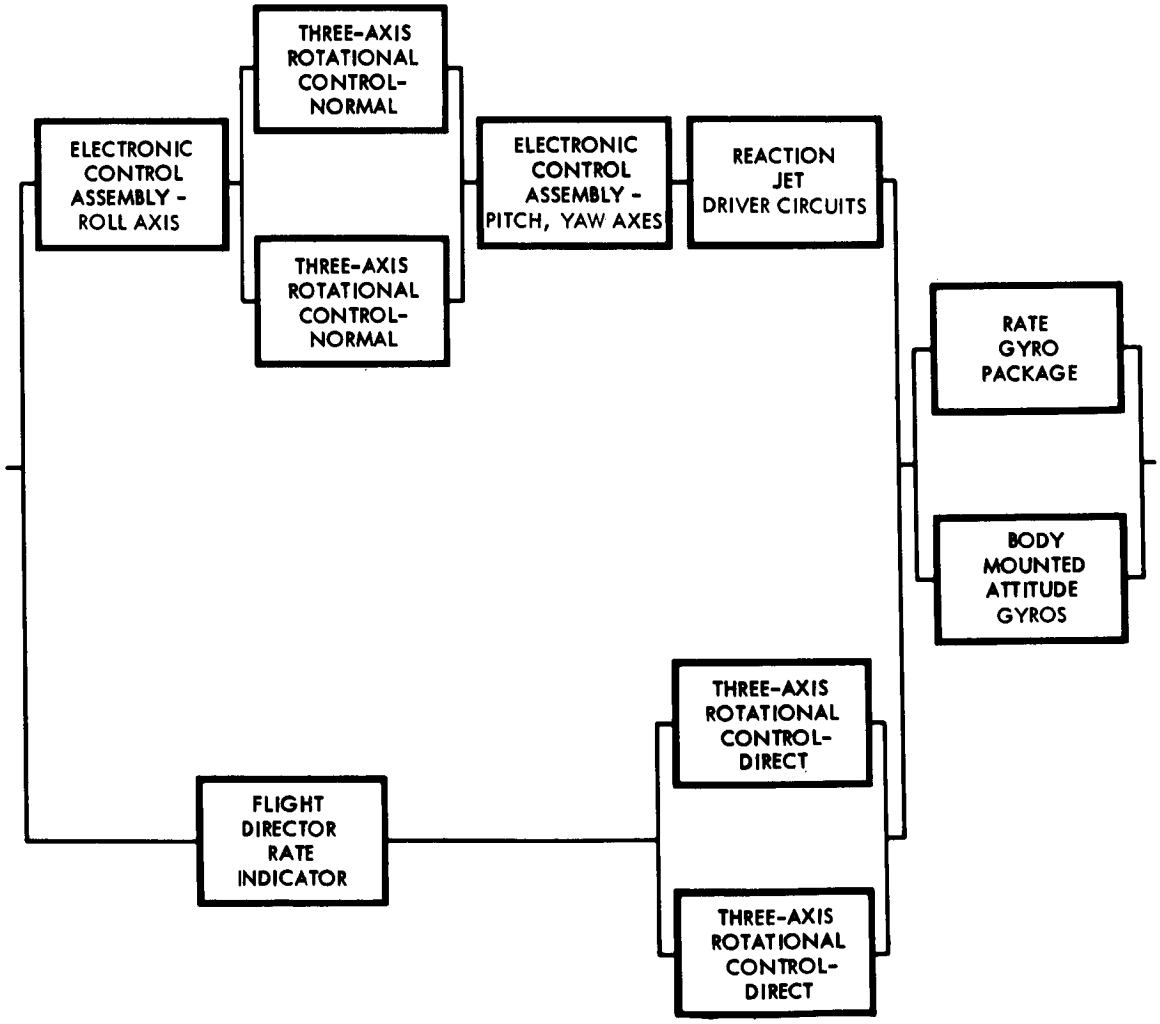
THREE-AXIS ROTATIONAL CONTROL-NORMAL
ELECTRONIC CONTROL ASSEMBLY - ROLL AXIS
ELECTRONIC CONTROL ASSEMBLY - PITCH, YAW AXES
REACTION JET DRIVER CIRCUITS
THREE-AXIS ROTATIONAL CONTROL-NORMAL
RATE GYRO PACKAGE
BODY MOUNTED ATTITUDE GYROS
THREE-AXIS ROTATIONAL CONTROL-DIRECT
FLIGHT DIRECTOR RATE INDICATOR
THREE-AXIS ROTATIONAL CONTROL-DIRECT

Figure 4-3. Abort Attitude Maneuver

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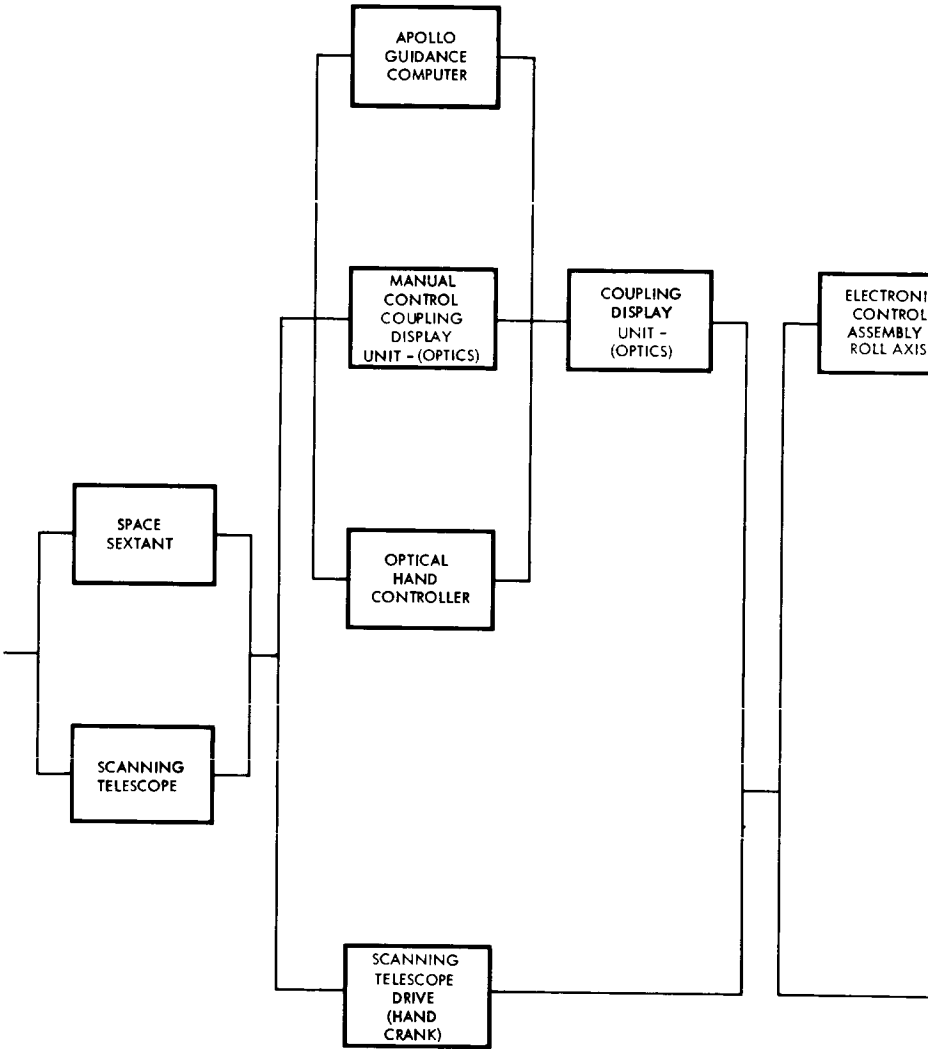

APOLLO GUIDANCE COMPUTER
MANUAL CONTROL COUPLING DISPLAY UNIT - (OPTICS)
COUPLING DISPLAY UNIT - (OPTICS)
ELECTRONI CONTROL ASSEMBLY ROLL AXIS
SPACE SEXTANT
OPTICAL HAND CONTROLLER
SCANNING TELESCOPE
SCANNING TELESCOPE DRIVE (HAND CRANK)

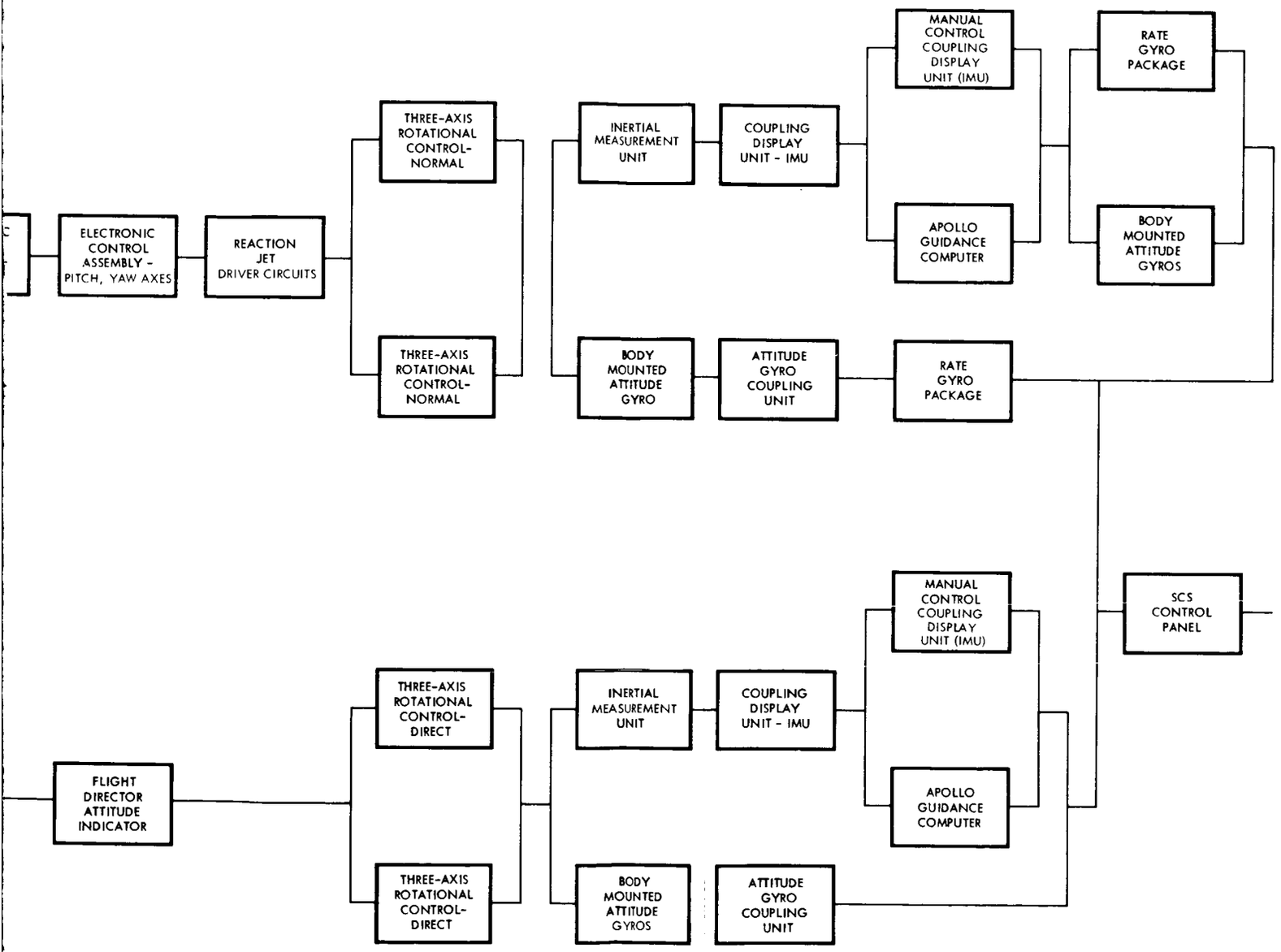
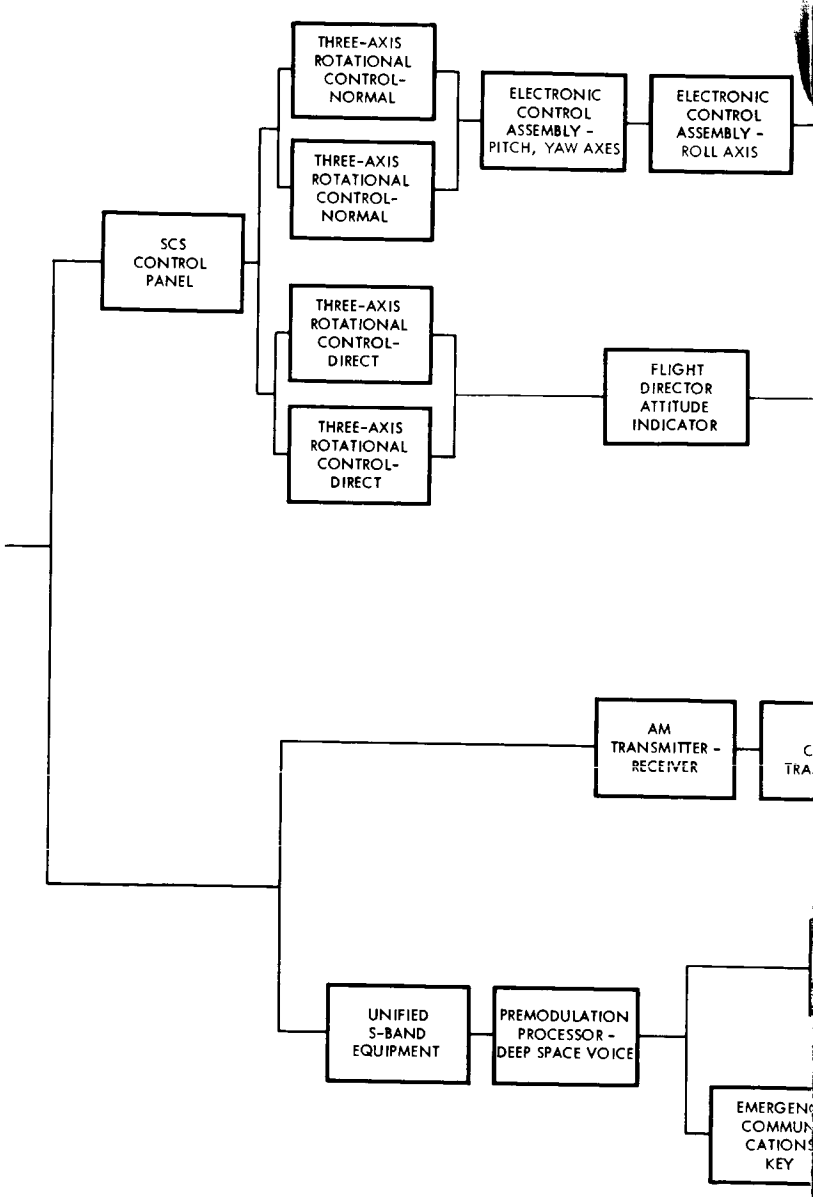


Figure 4-4. Platform Erection


THREE-AXIS ROTATIONAL CONTROL-NORMAL
THREE-AXIS ROTATIONAL CONTROL-NORMAL
ELECTRONIC CONTROL ASSEMBLY - PITCH, YAW AXES
ELECTRONIC CONTROL ASSEMBLY - ROLL AXIS
SCS CONTROL PANEL
THREE-AXIS ROTATIONAL CONTROL-DIRECT
THREE-AXIS ROTATIONAL CONTROL-DIRECT
FLIGHT DIRECTOR ATTITUDE INDICATOR
AM TRANSMITTER - RECEIVER
UNIFIED S-BAND EQUIPMENT
PREMODULATION PROCESSOR - DEEP SPACE VOICE

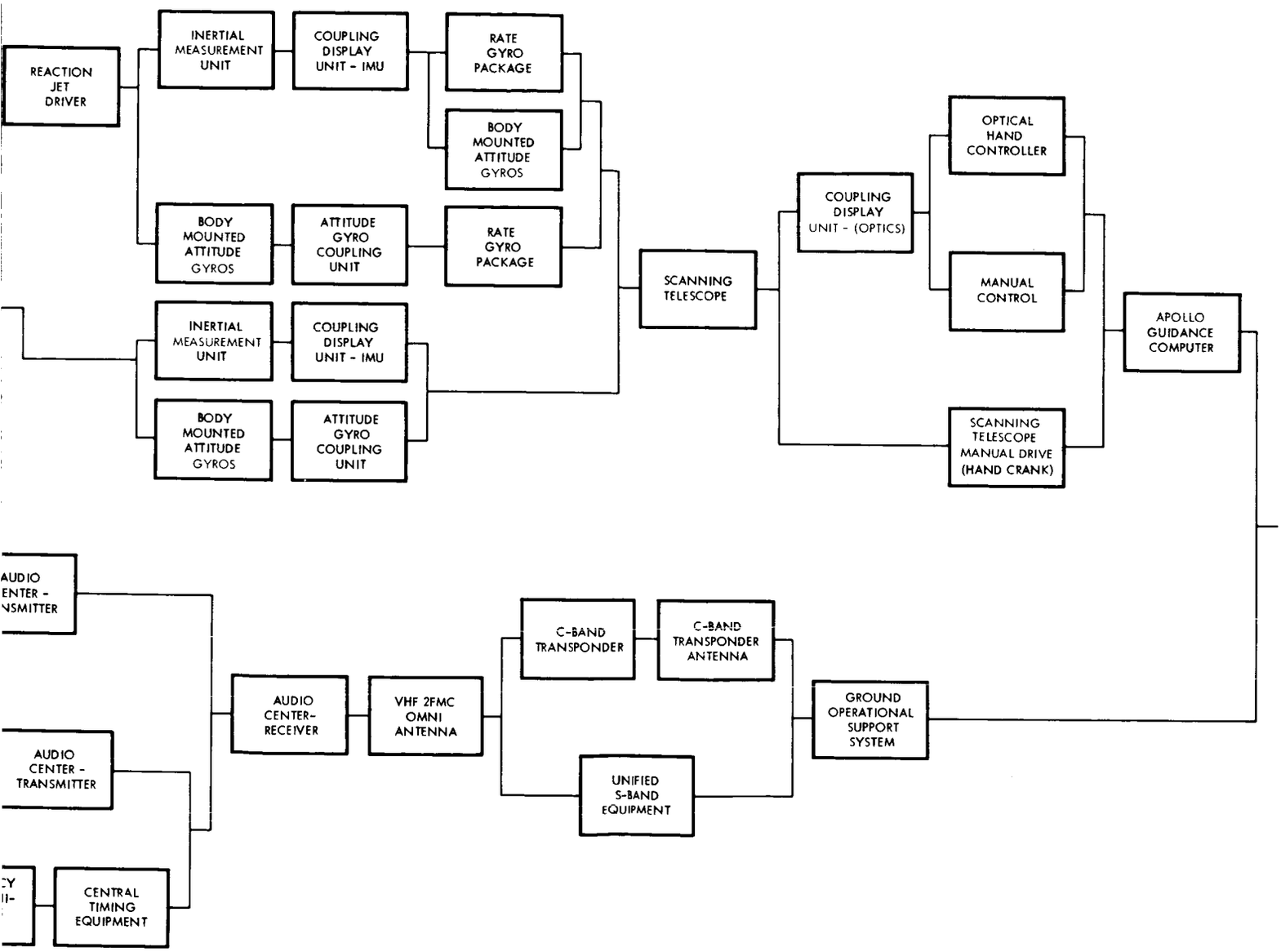


Figure 4-5. Navigational Sighting (Cisearth)

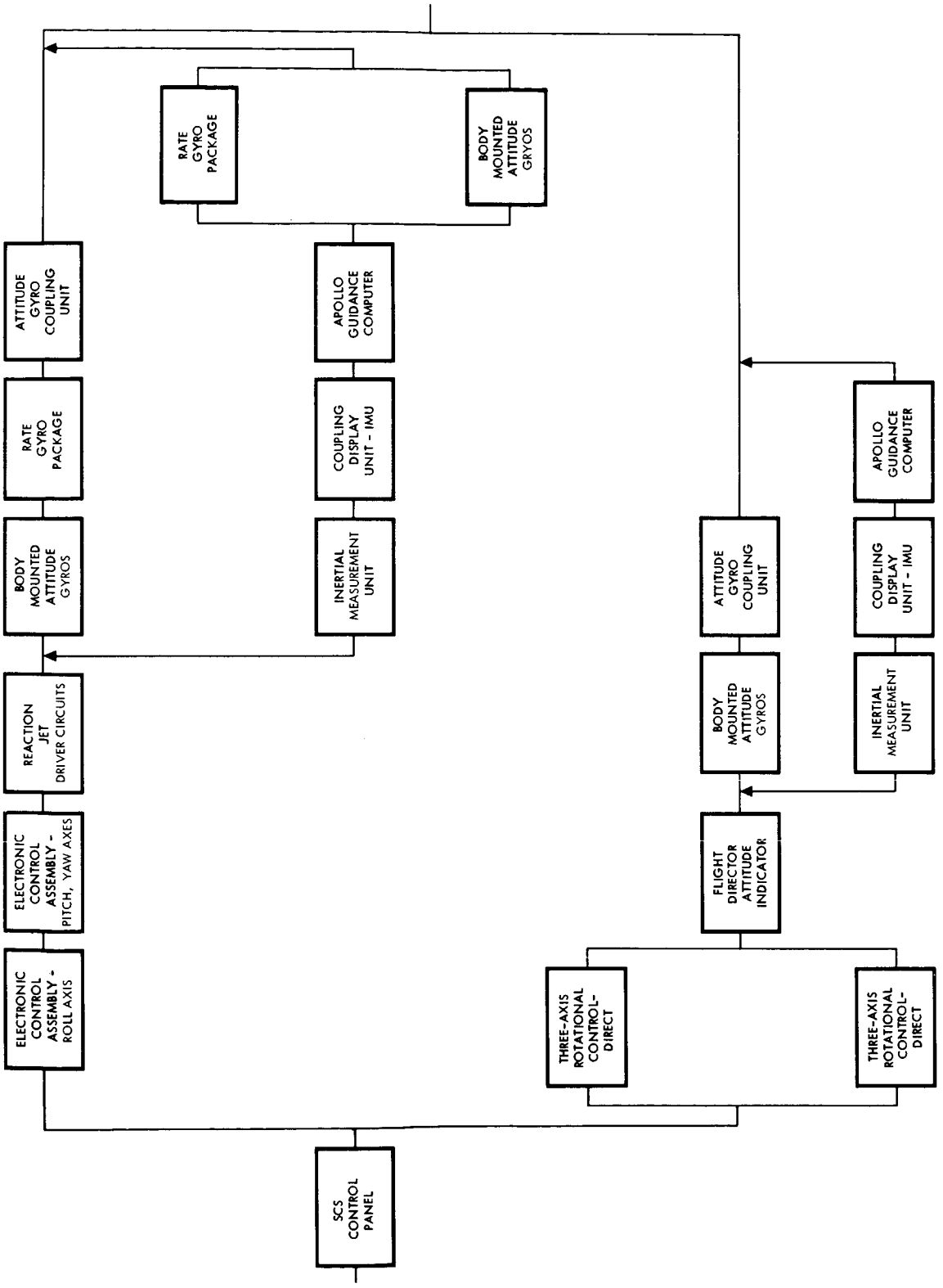


Figure 4-6. Dwell and Coast

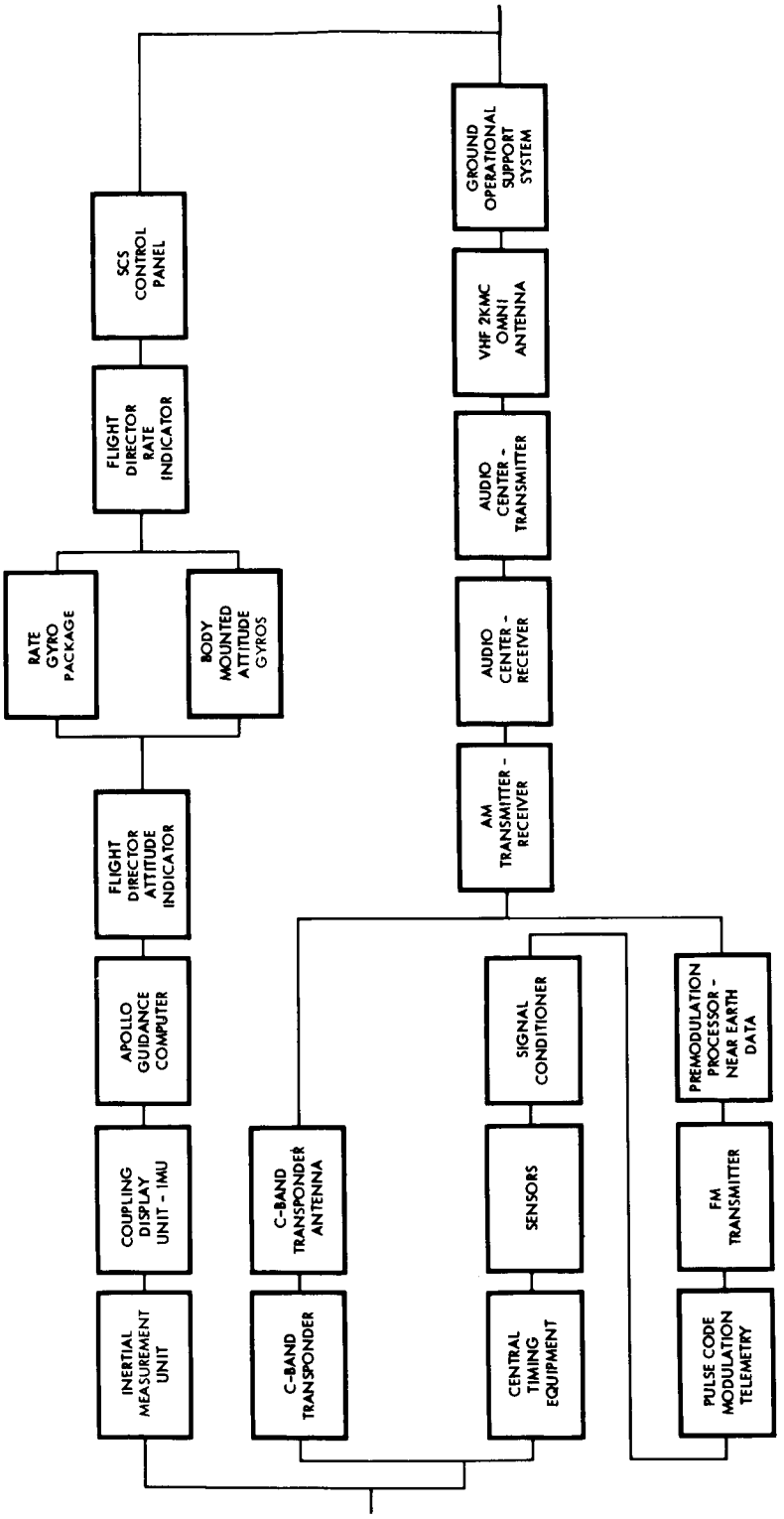


Figure 4-7. Translunar Inject

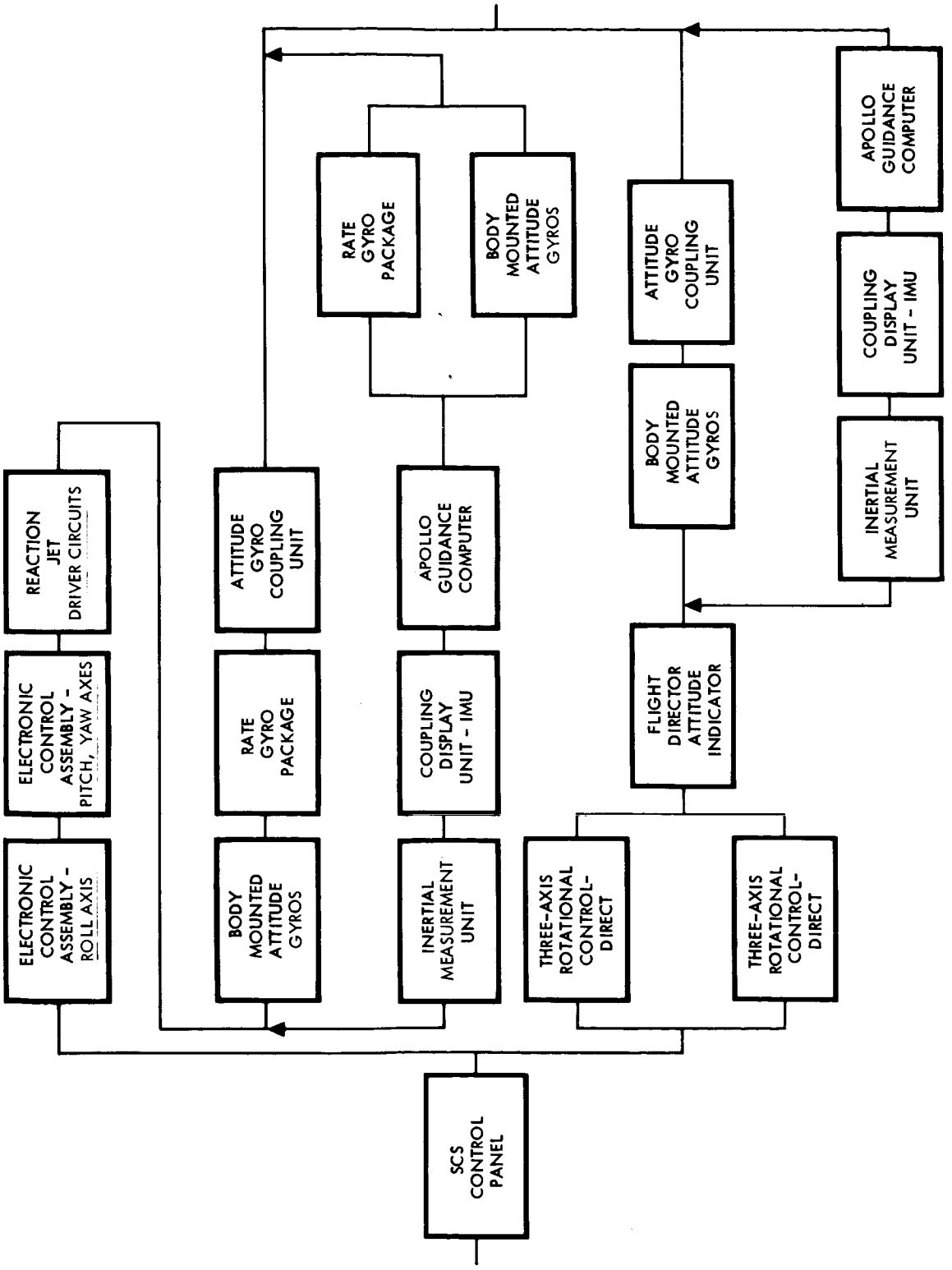
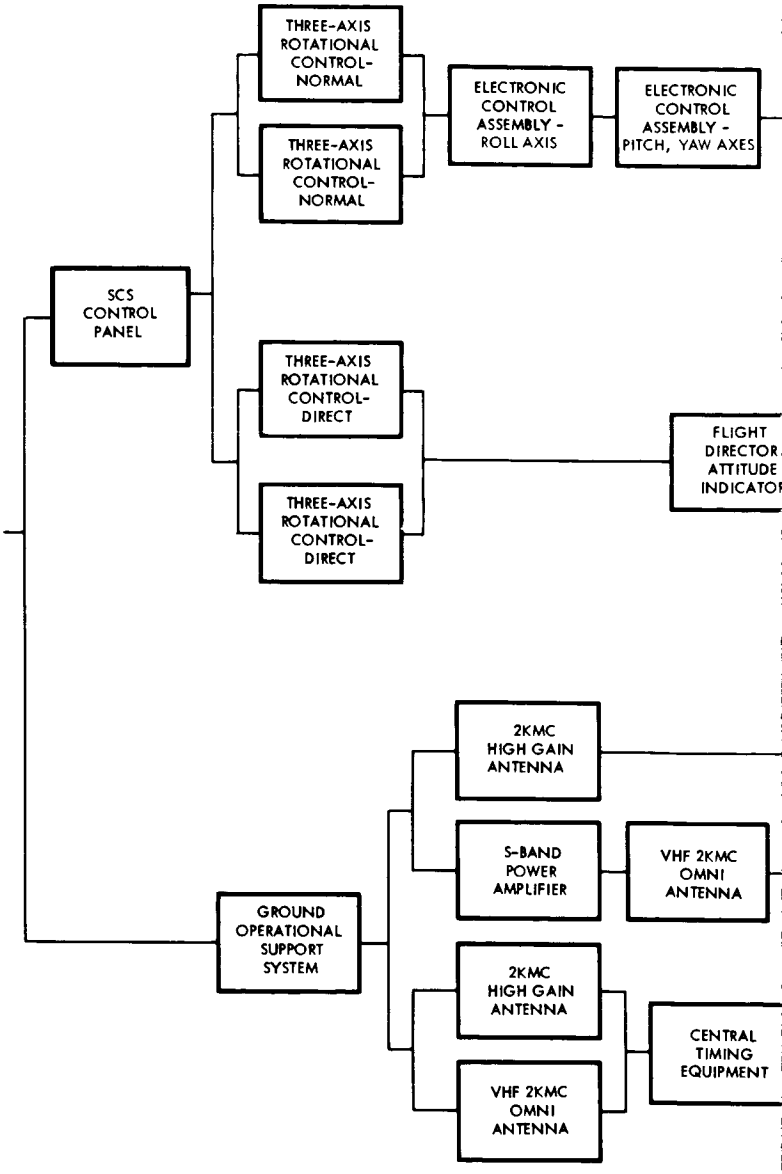


Figure 4-8. Translunar Coast Inner-Radiation Belt


SCS CONTROL PANEL
THREE-AXIS ROTATIONAL CONTROL-NORMAL
THREE-AXIS ROTATIONAL CONTROL-NORMAL
ELECTRONIC CONTROL ASSEMBLY - ROLL AXIS
ELECTRONIC CONTROL ASSEMBLY - PITCH, YAW AXES
THREE-AXIS ROTATIONAL CONTROL-DIRECT
THREE-AXIS ROTATIONAL CONTROL-DIRECT
FLIGHT DIRECTOR ATTITUDE INDICATOR
GROUND OPERATIONAL SUPPORT SYSTEM
2KMC HIGH GAIN ANTENNA
S-BAND POWER AMPLIFIER
VHF 2KMC OMNI ANTENNA
2KMC HIGH GAIN ANTENNA
VHF 2KMC OMNI ANTENNA
CENTRAL TIMING EQUIPMENT

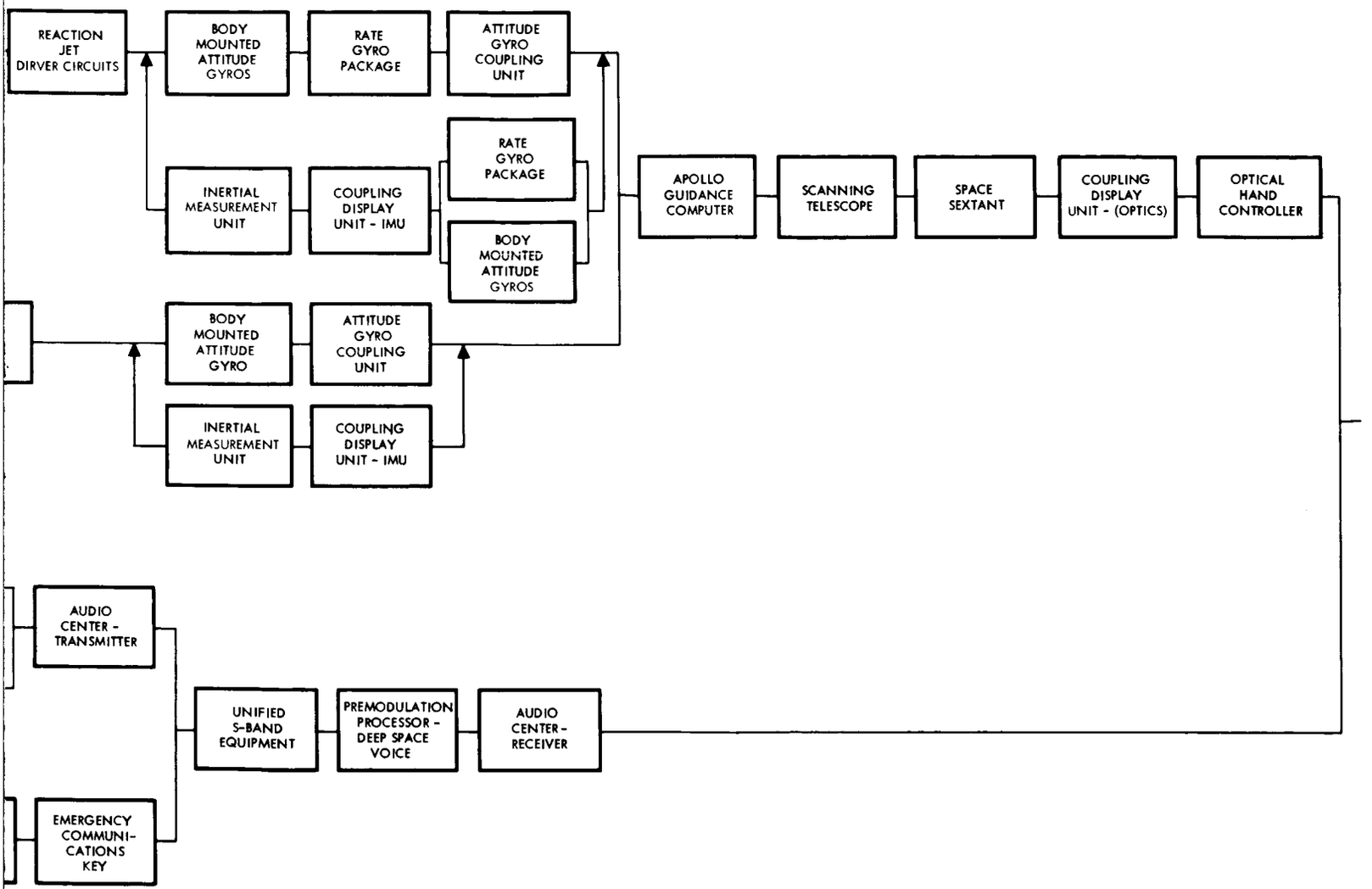


Figure 4-9. Navigational Sighting, Translunar and Transearth

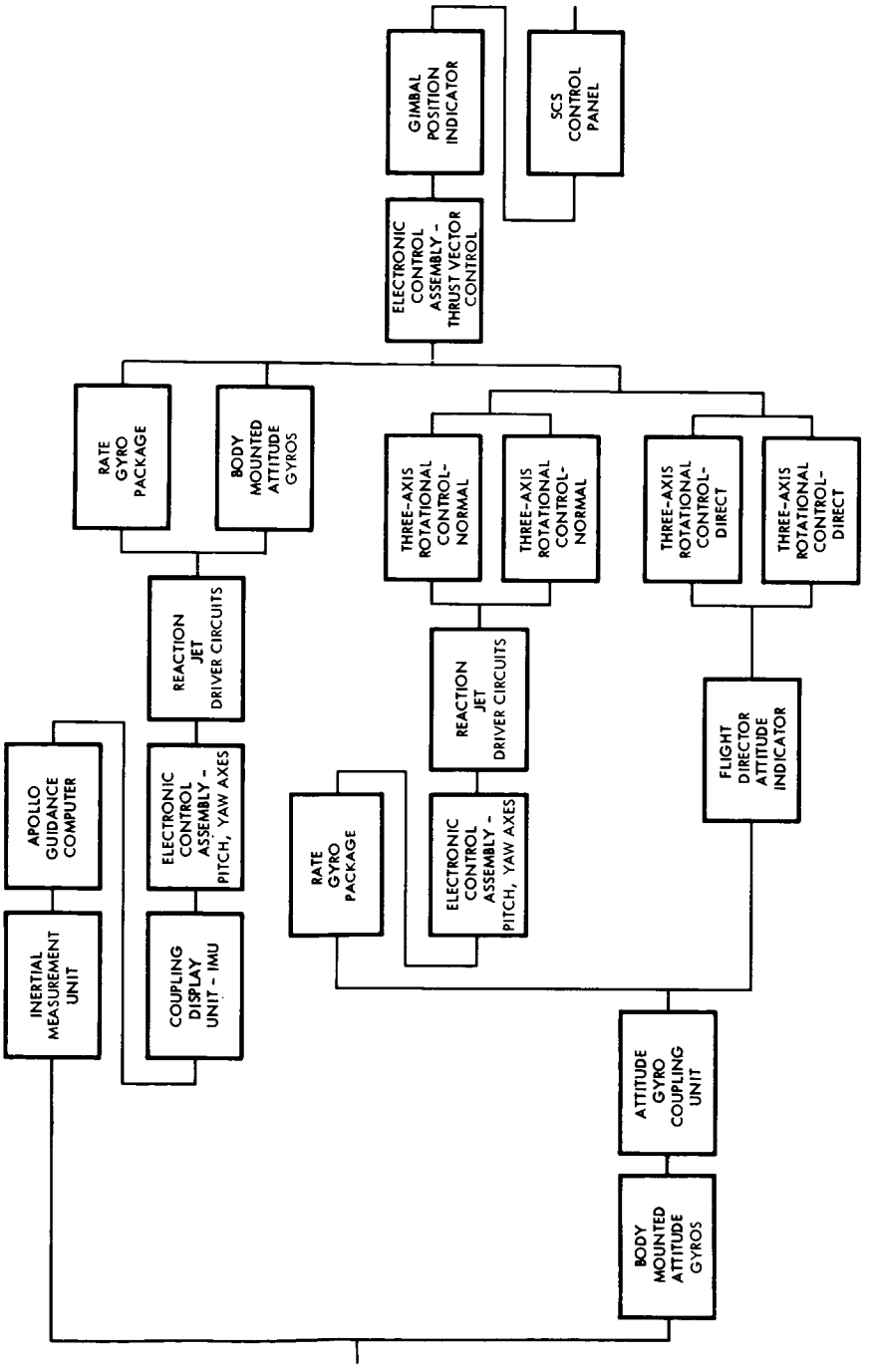

GIMBAL POSITION INDICATOR
SCS CONTROL PANEL
ELECTRONIC CONTROL ASSEMBLY - THRUST VECTOR CONTROL
RATE GYRO PACKAGE
BODY MOUNTED ATTITUDE GYROS
THREE-AXIS ROTATIONAL CONTROL-NORMAL
THREE-AXIS ROTATIONAL CONTROL-NORMAL
THREE-AXIS ROTATIONAL CONTROL-DIRECT
THREE-AXIS ROTATIONAL CONTROL-DIRECT
REACTION JET DRIVER CIRCUITS
REACTION JET DRIVER CIRCUITS
FLIGHT DIRECTOR ATTITUDE INDICATOR
APOLLO GUIDANCE COMPUTER
ELECTRONIC CONTROL ASSEMBLY - PITCH, YAW AXES
RATE GYRO PACKAGE
ELECTRONIC CONTROL ASSEMBLY - PITCH, YAW AXES
INERTIAL MEASUREMENT UNIT
COUPLING DISPLAY UNIT - IMU
ATTITUDE GYRO COUPLING UNIT
BODY MOUNTED ATTITUDE GYROS

Figure 4-10. Thrust Vector Alignment

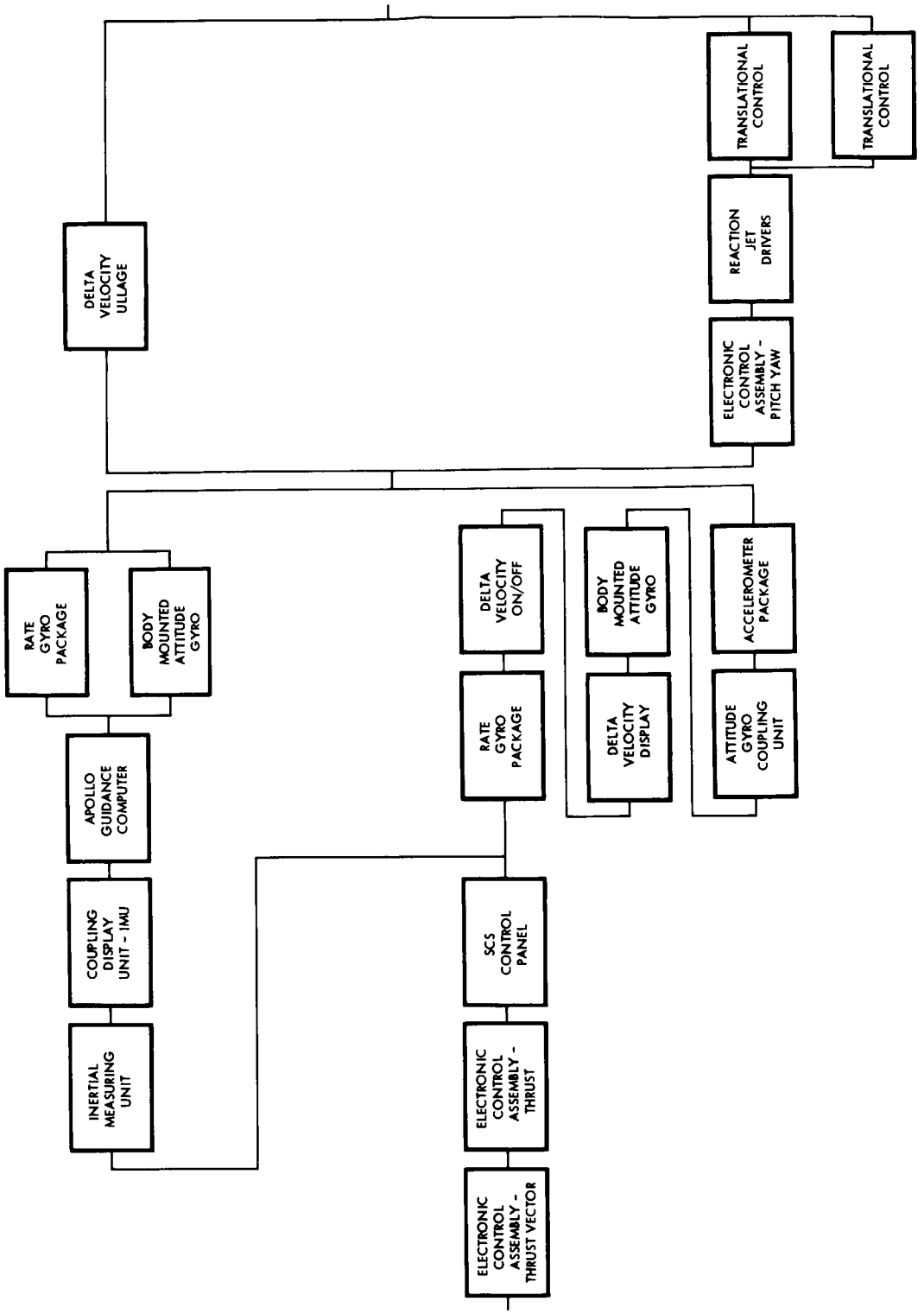
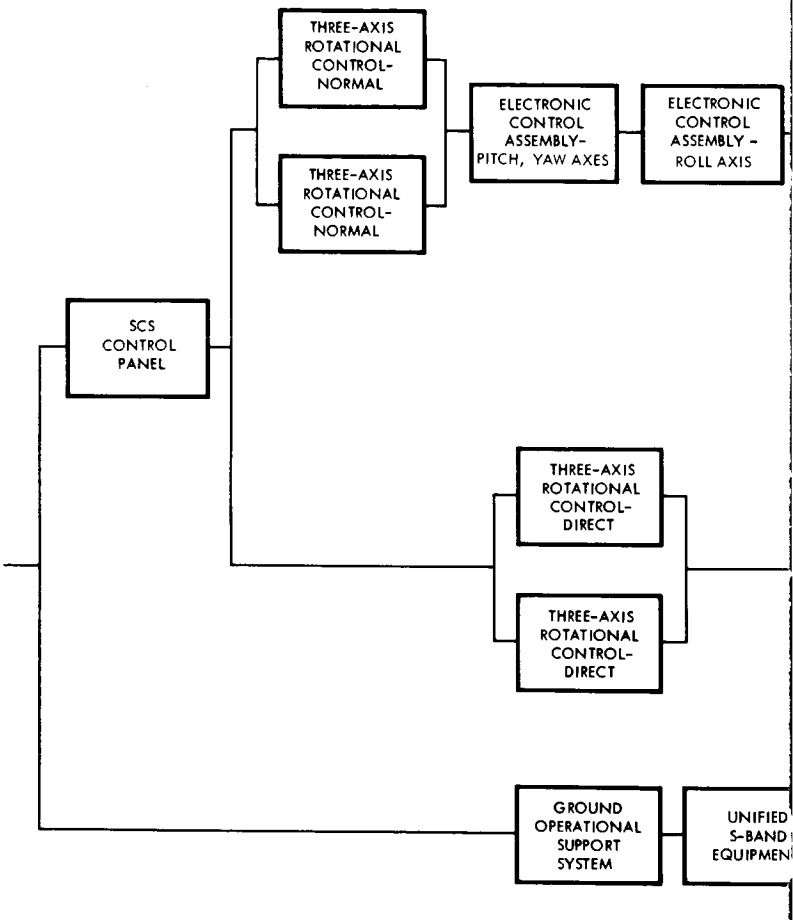


Figure 4-11. ΔV Firing


THREE-AXIS ROTATIONAL CONTROL-NORMAL
THREE-AXIS ROTATIONAL CONTROL-NORMAL
ELECTRONIC CONTROL ASSEMBLY-PITCH, YAW AXES
ELECTRONIC CONTROL ASSEMBLY - ROLL AXIS
SCS CONTROL PANEL
THREE-AXIS ROTATIONAL CONTROL-DIRECT
THREE-AXIS ROTATIONAL CONTROL-DIRECT
GROUND OPERATIONAL SUPPORT SYSTEM
UNIFIED S-BAND EQUIPMEN

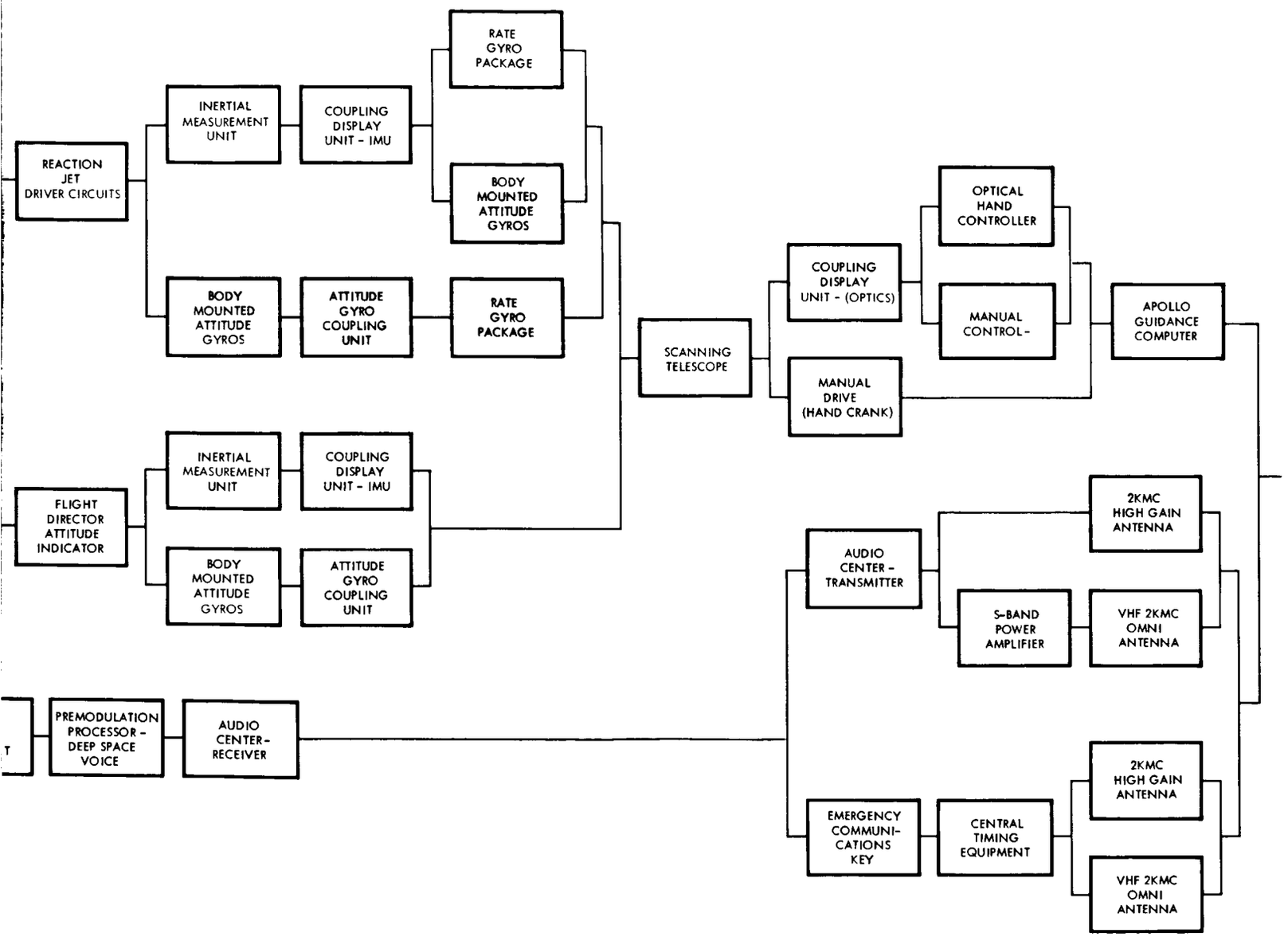


Figure 4-12. Navigational Sighting, Antelunar

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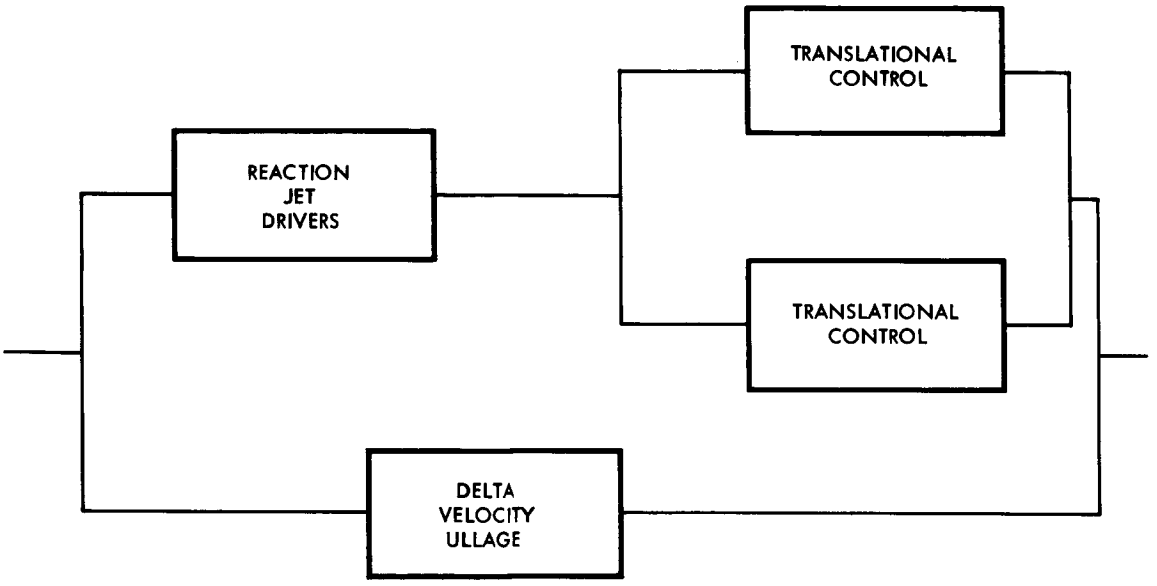


Figure 4-13. Service Module Separation

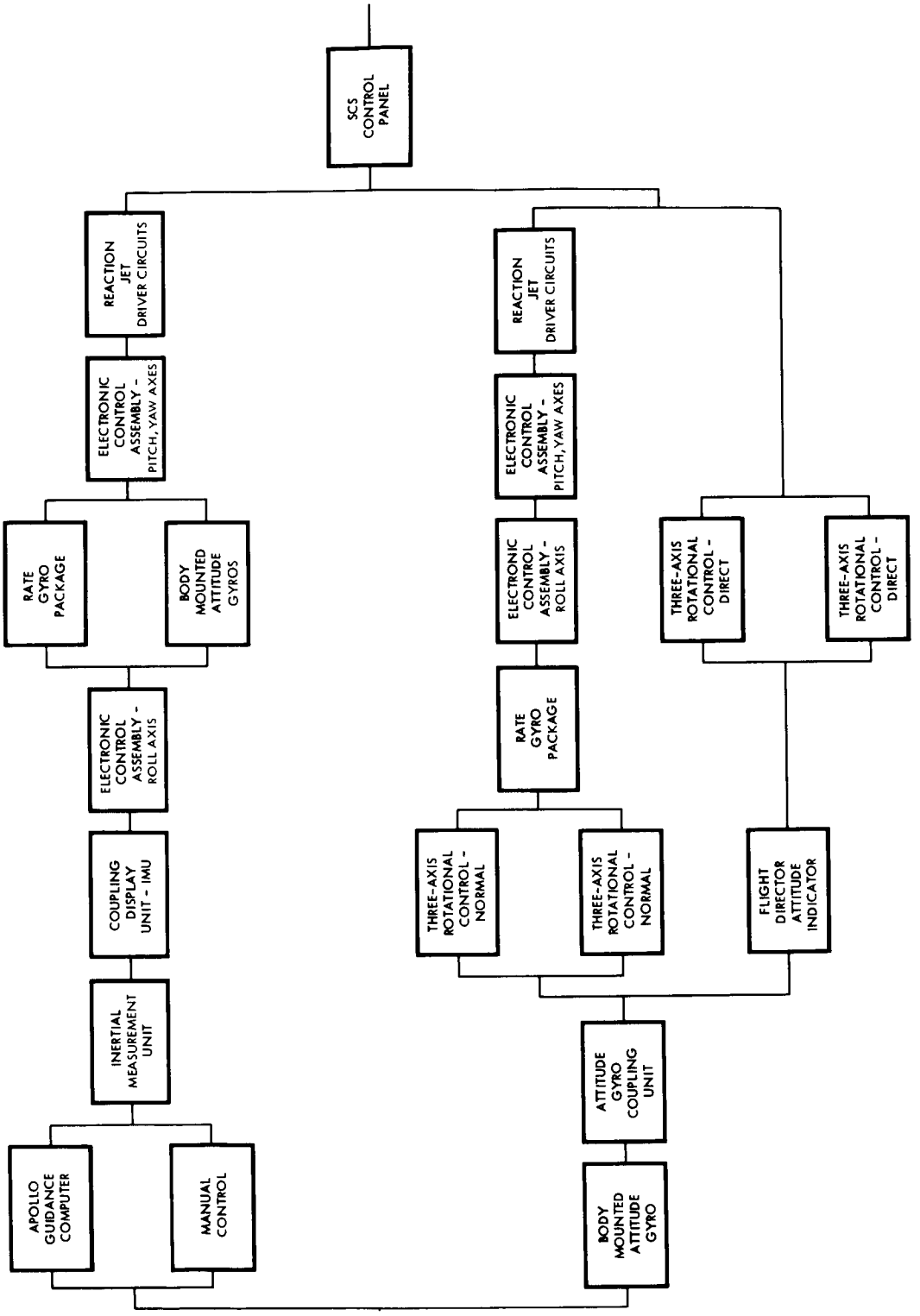

SCS CONTROL PANEL
REACTION JET DRIVER CIRCUITS
ELECTRONIC CONTROL ASSEMBLY - PITCH, YAW AXES
RATE GYRO PACKAGE
BODY MOUNTED ATTITUDE GYROS
ELECTRONIC CONTROL ASSEMBLY - ROLL AXIS
COUPLING DISPLAY UNIT - IMU
INERTIAL MEASUREMENT UNIT
APOLLO GUIDANCE COMPUTER
MANUAL CONTROL
REACTION JET DRIVER CIRCUITS
ELECTRONIC CONTROL ASSEMBLY - PITCH, YAW AXES
ELECTRONIC CONTROL ASSEMBLY - ROLL AXIS
RATE GYRO PACKAGE
THREE-AXIS ROTATIONAL CONTROL - NORMAL
THREE-AXIS ROTATIONAL CONTROL - NORMAL
ATTITUDE GYRO COUPLING UNIT
BODY MOUNTED ATTITUDE GYRO
THREE-AXIS ROTATIONAL CONTROL - DIRECT
THREE-AXIS ROTATIONAL CONTROL - DIRECT
FLIGHT DIRECTOR ATTITUDE INDICATOR

Figure 4-14. Orientation for Reentry

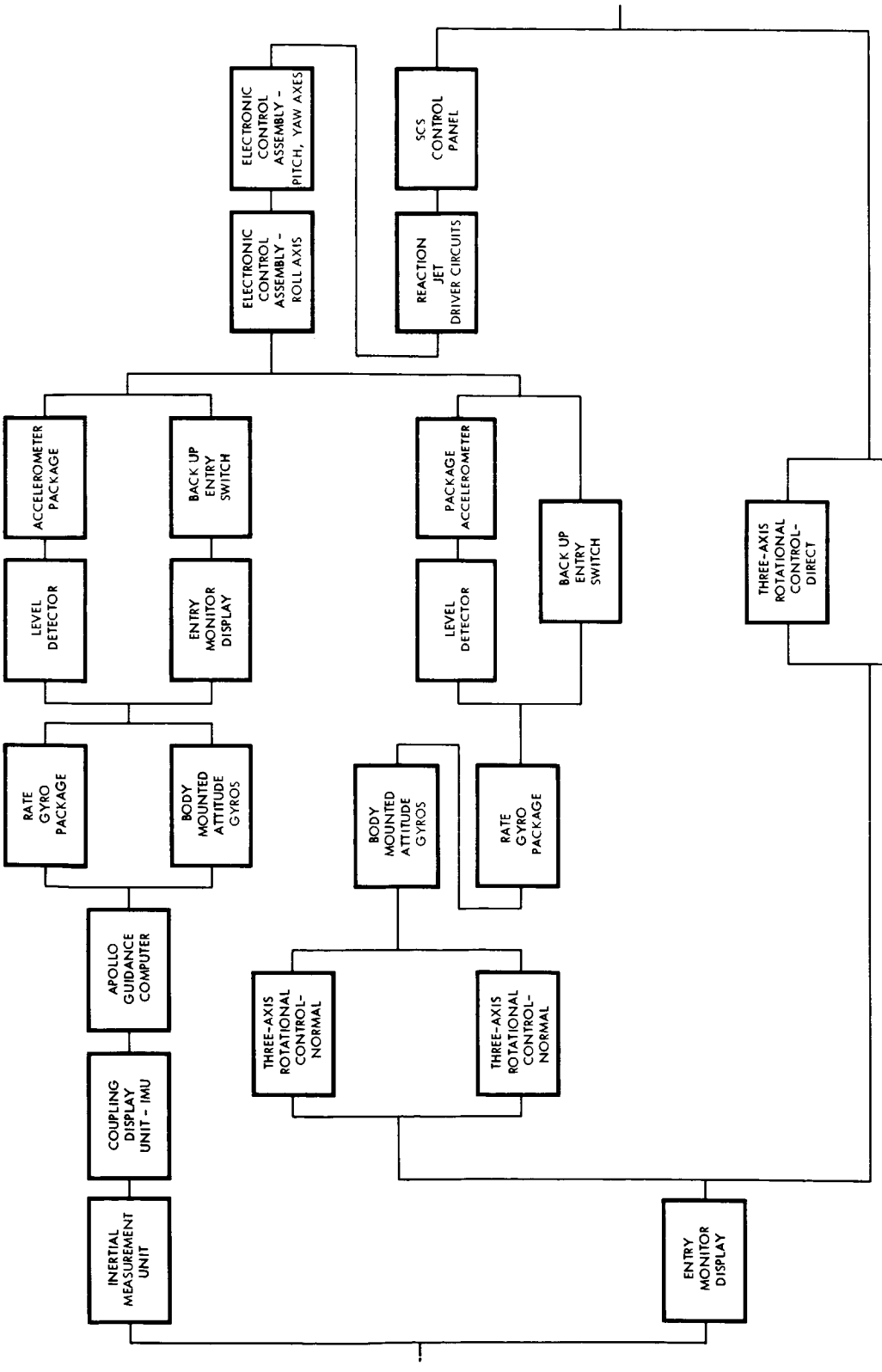


Figure 4-15. Entry

Figure 4-16. Drogue Chute Deployment to Landing

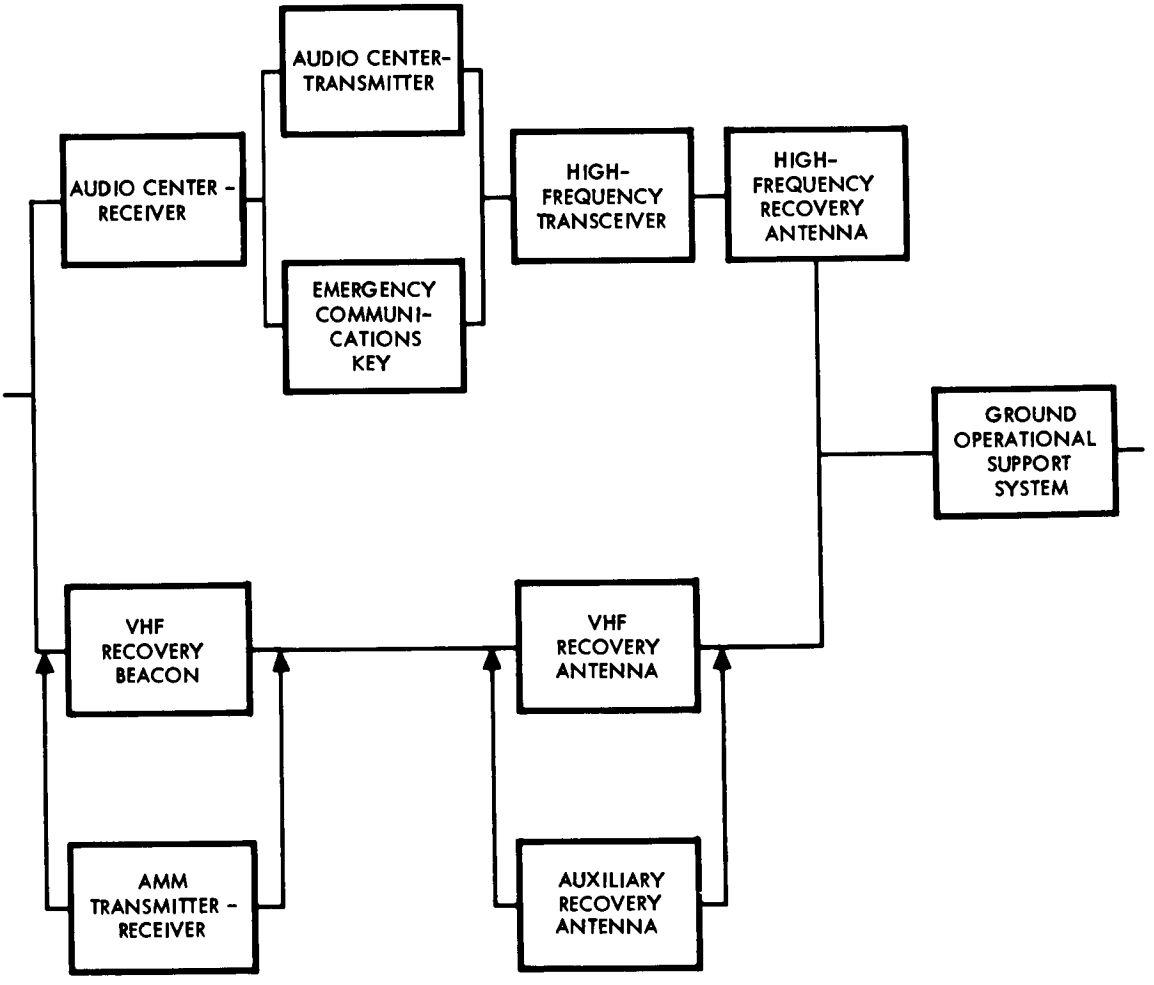
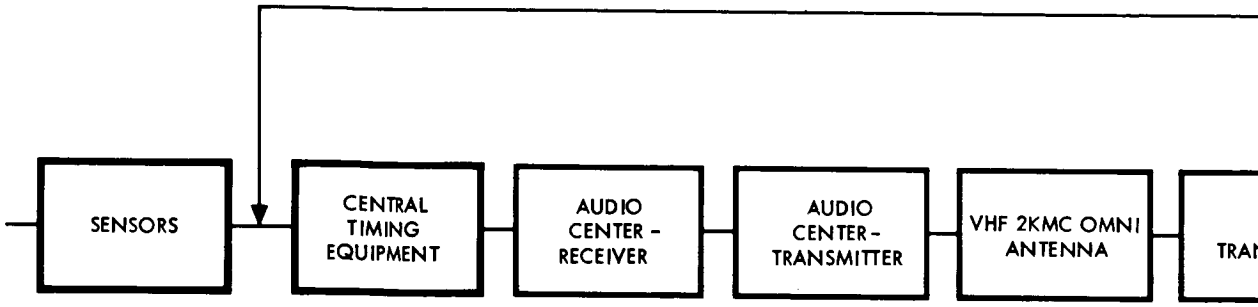
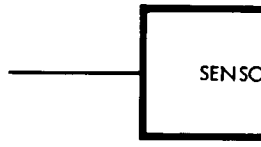
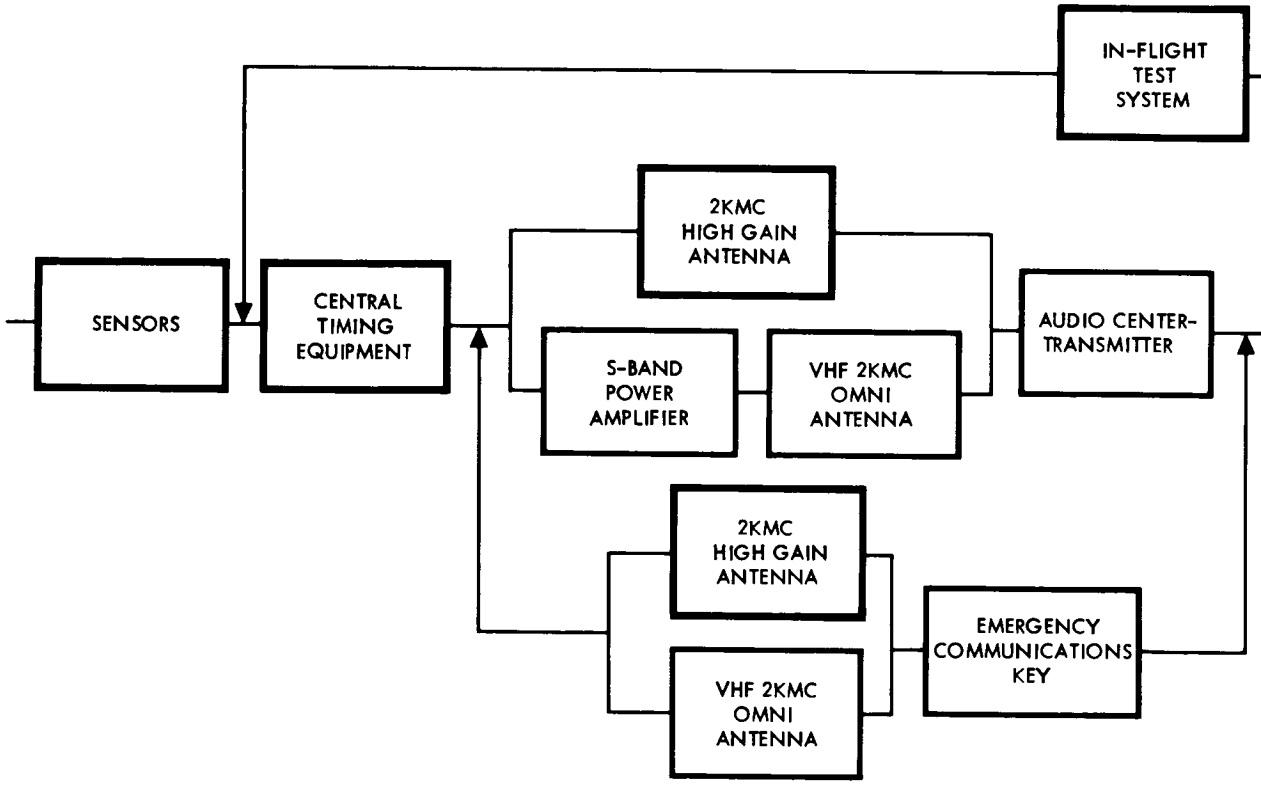


Figure 4-17. Post Landing





TRANSLUNAR, ANTELUNAR, &



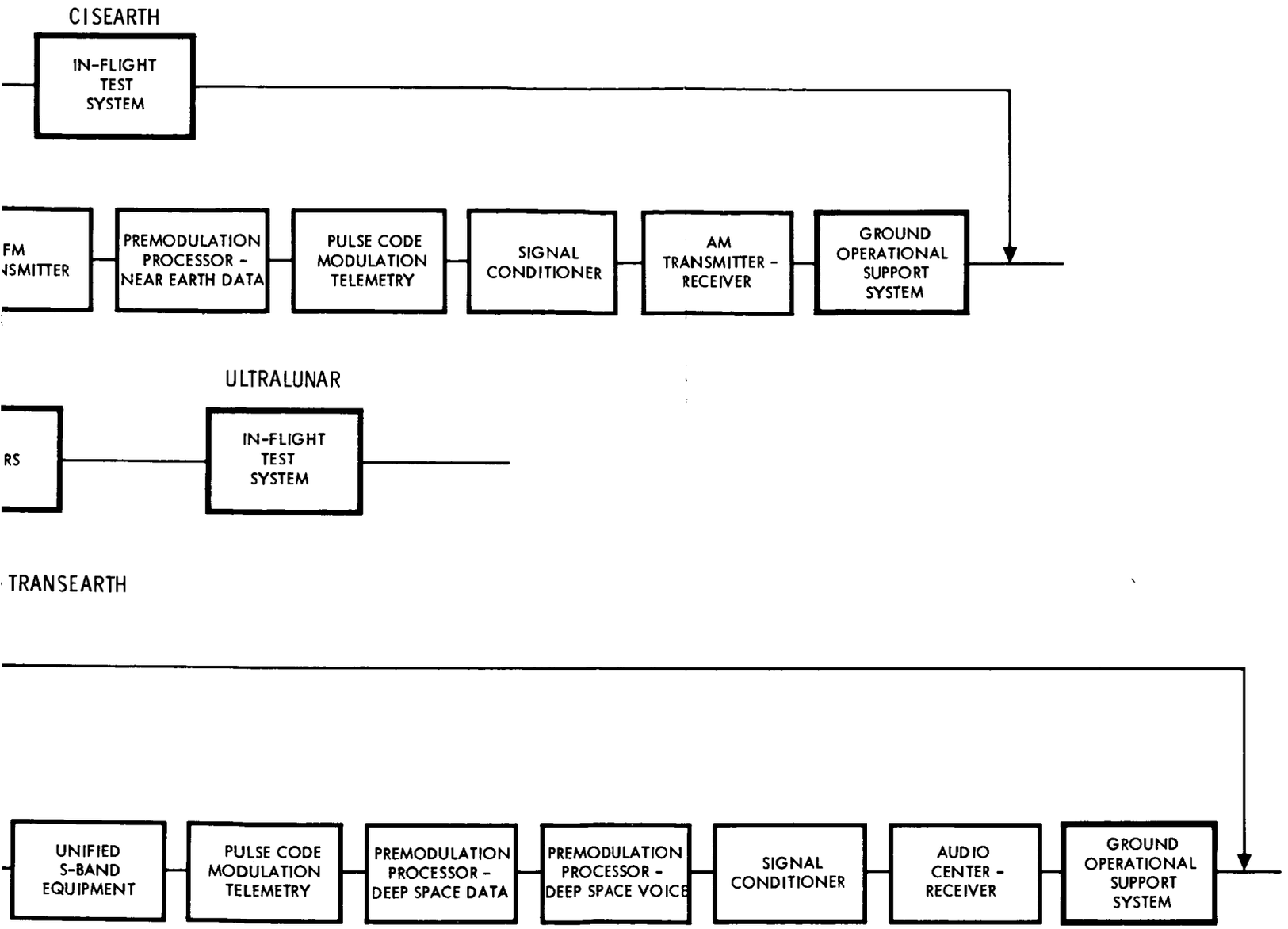


Figure 4-18. Monitor Logic for Sparing

[redacted]

COMMUNICATIONS AND DATA

SUMMARY

The communications and data subsystem reliability estimates indicate that the requirements will be met with minimal sparing. Effort to maximize the reliability of the subsystem design by performing configuration studies in support of the design effort is continuing. The test plans and procedures of the subcontractor are being monitored to maximize the value of the qualification tests. The logic interactions with the other subsystems of the Spacecraft and subsystems of the lunar excursion module are under continuous investigation for possible improvement.

ANALYSIS

The current mission success reliability predictions analytically determined for the equipment of the communications and data subsystem are given in Table 4-1. No sparing or redundancy is considered in these predictions.

Failure-Effects Analysis

The results of a preliminary failure-effects analysis of the communications and data subsystem at the equipment level are given in Table 4a-K-12 at the end of this section.

C-Band Transponder

A reliability evaluation of the C-band transponder equipment was based on design information in ACF's technical proposal using one spare magnetron. Figure 4-19 shows the reliability logic diagram for the transponder with the failure rates for the various subassemblies based upon Minuteman parts failure rates. Results of analysis indicate that the design submitted was capable of achieving a success probability of 0.99992, which exceeds the apportioned reliability of 0.9999. Additional sparing, therefore, will not be required if high-reliability parts are used and the inherent reliability is not degraded in manufacturing, checkout, shipping, etc.

DSIF High-Gain Antenna

Several Deep Space Instrumentation Facility high-gain antenna configurations were investigated to determine the feasibility of achieving the

apportioned reliability of 0.9999. The results shown in Table 4-2 indicate that with minimum parts improvement a redundant gimbal drive can meet the apportionment.

Table 4-1. Communications and Data Subsystem Apportionment and Predictions

Equipment	Mission Success Reliability	
	Apportioned	Predicted
VHF recovery beacon	0.99992	0.99981
HF tranceiver	0.9999	0.999935
VHF/AM transmitter-receiver	0.99996	0.99904
VHF/FM transmitter	0.99997	0.99987
Audio center	0.9995	0.9953
Data storage	0.993	0.993
C-band transponder	0.9999	0.99938
Unified S-band equipment	0.999	0.9930
Unified S-band power amplifier	0.9992	0.9914
Controls and displays	0.9991	0.9980
Premodulation processor	0.9995	0.9942
Signal conditioner	0.9996	0.9973
PCM telemetry	0.997	0.937
VHF/2 kmc omni antenna	0.999998	0.999998

Signal Conditioner

An estimate has been made of the signal conditioner equipment reliability to determine the feasibility of attaining the apportioned reliability of 0.9996. Based on Minuteman parts failure rates, 0.9992 reliability was found achievable. This "worst-case" prediction (assuming no redundancy at lower levels of assembly) is extremely pessimistic, since the 117 signal

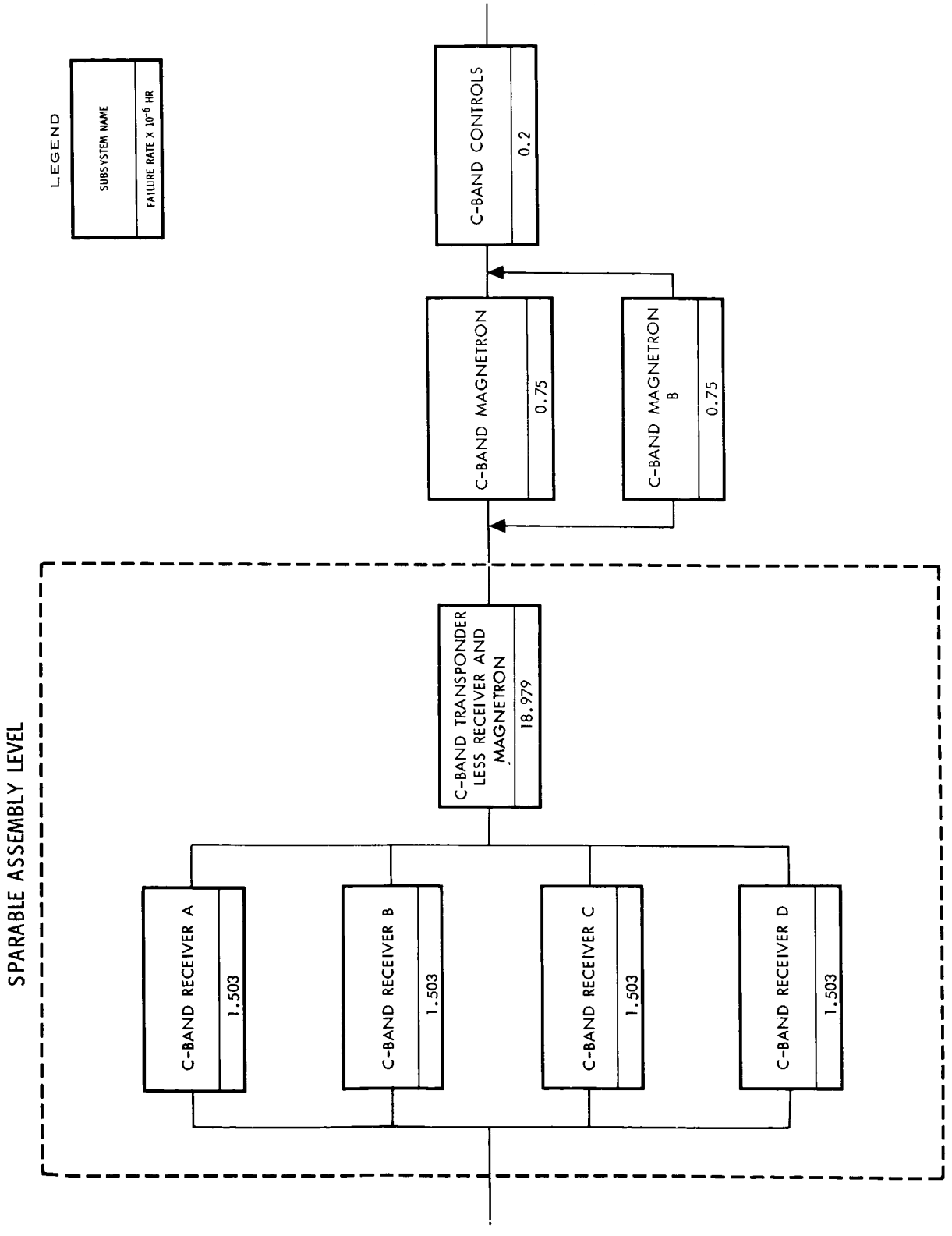


Figure 4-19. C-Band Transponder

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Table 4-2. DSIF High-Gain Antenna Reliability Comparisons

	Mission Success Reliability	
Configuration	Apportioned	Predicted
No redundancy	0.9999	0.996300
Redundant gimbal drive	0.9999	0.999886
Complete redundancy	0.9999	0.999987

conditioners are not in series with each other. Therefore, relatively little difficulty will be encountered in meeting the over-all requirement.

VHF/AM Transmitter-Receiver

A reliability evaluation of the VHF/AM transmitter-receiver has been based upon the latest detailed functional information. Figure 4-20 shows the reliability logic diagram for the transmitter-receiver with the failure rates for the various subassemblies based upon Minuteman parts failure rates. Results of analysis indicate that the design evaluated is capable of achieving a mission success probability of 0.99917. The apportioned reliability for this equipment is 0.99996; therefore, sparing will be necessary to achieve the apportioned value.

C-Band Transponder Power Supply

A configuration study has been completed to compare the reliability of the proposed DC power supply and the existing AC power supply utilized in the C-band transponder of the communications and data subsystem. The existing power supply utilizes the AC output from the DC to AC converter in the Apollo command module. The proposed DC power supply would utilize the output directly from the DC buses. The configuration in Figure 4-21 and a duty cycle of 9.7 hours were used in the calculations. The results obtained are shown in Table 4-3.

Since it was found that the proposed change for the internal power supply of the C-band transponder will not cause a significant change in equipment reliability, primary consideration was recommended for other design parameters.

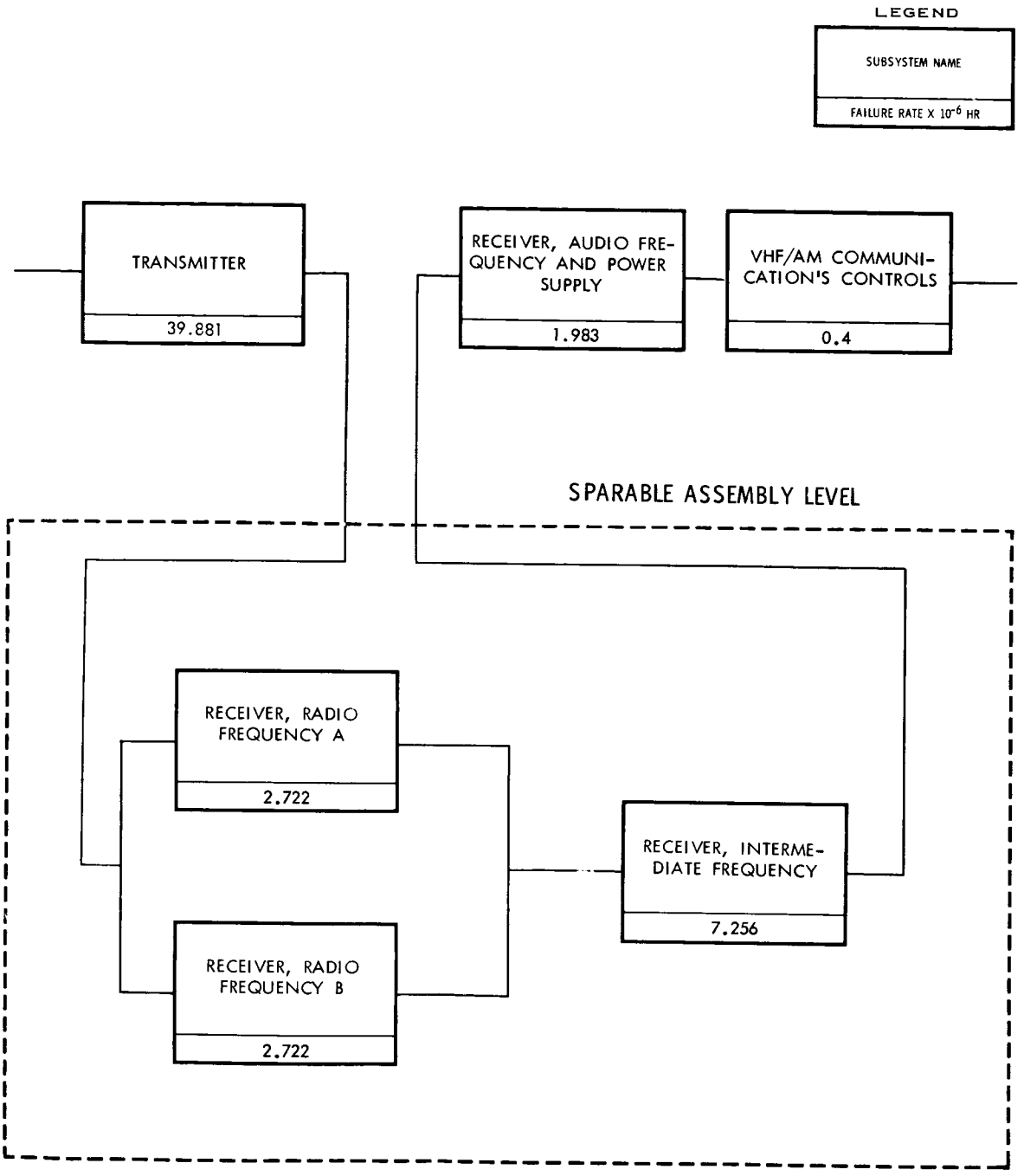

Figure 4-20. VHF/AM Transmitter-Receiver

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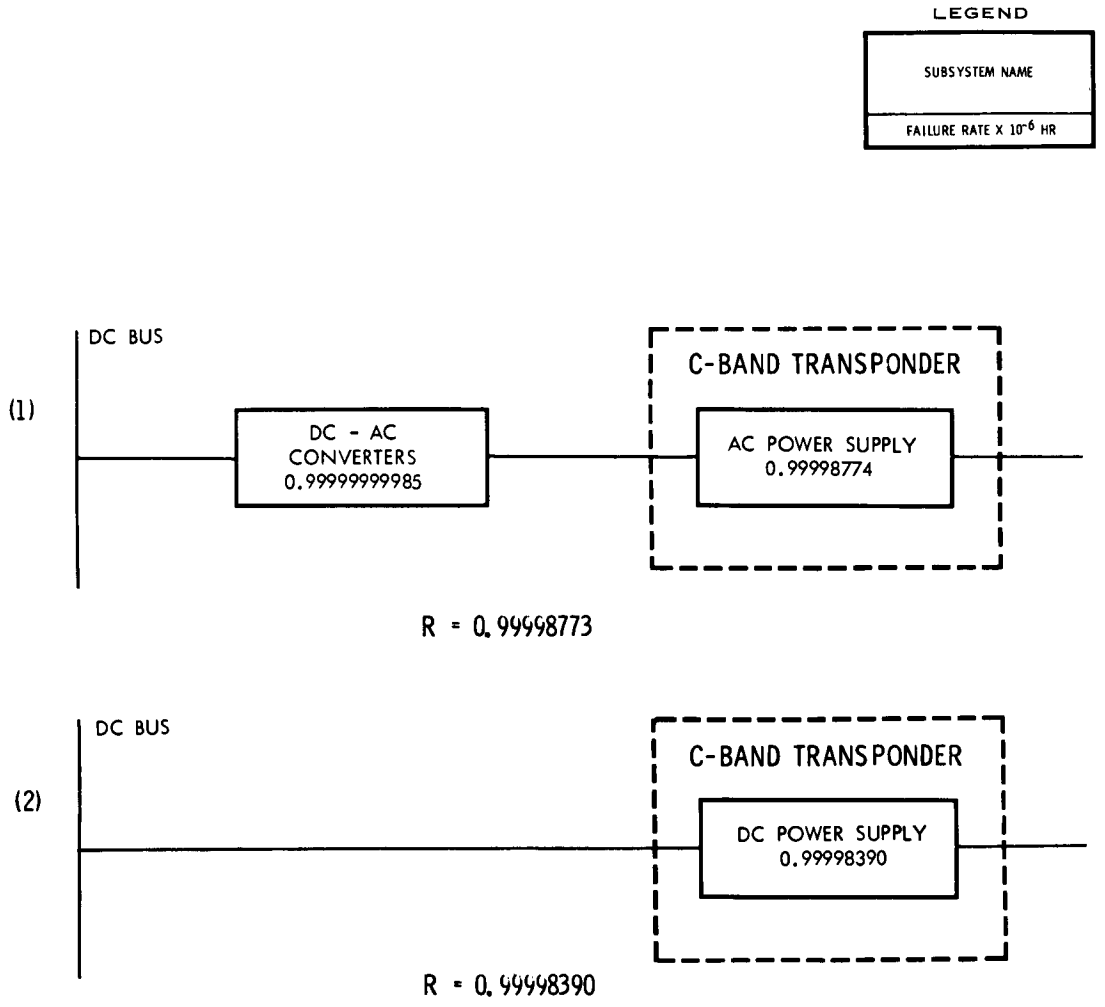


Figure 4-21. Configurations

Table 4-3. C-Band Transponder Power Supply Reliability Comparisons

Configuration	Mission Success Reliability
(1) A-c power supply and dc-ac converters	0.99998773
(2) D-c power supply	0.99998390

SUBSYSTEM DESCRIPTION

Figure 4-22 shows the expanded reliability logic diagram of the equipment in the communications and data and instrumentation subsystems used as an alternate mode to equipment of the guidance and navigation and stabilization and control subsystems necessary for performing a navigational sighting in the deep-space phases of the LOR mission. The communication and data subsystem subassemblies are shown in solid lines, the instrumentation subsystems subassemblies in dotted lines.

The equipment of the communications and data subsystem are used to complement the tracking and ranging function and to supply on-board capability for voice communications with the earth. Thus, tracking and ranging information can be relayed from the earth to the spacecraft via voice communications giving a navigational sighting.

The voice capability is supplied by the unified S-band equipment (USB) and associated components. Voice transmission is accomplished in the following manner: The audio voice signal is amplified by audio center equipment amplifiers (ACTAA, ACTAB, ACTAC) in conjunction with the audio control centers (ACCUA, ACCUB, ACCUC). There is an audio center for each crew member. The proper mode of transmission is selected by means of switching relays (PDVR) located in the premodulation processor and associated controls (PDVC).

The voice signal is then modulated and amplified by the unified S-band equipment (either the primary unified S-band transponder, USBTA, or its standby, USBTB, and the associated controls, USBC). The output is fed to either the 2KMC high-gain antenna (HGA) or the 2KMC omni portion of the VHF/2KMC omni antenna (DSC) in conjunction with the S-band power amplifier, which comprises a traveling-wave tube (SPTWA) and its alternate (SPTWB), a power supply (SPAPA) and its alternate (SPAPB), and associated controls (SPAC). The S-band power amplifier is necessary to allow the DSC to radiate satisfactorily at lunar distances.

The audio link of the transmission function may also be accomplished by the emergency communications key (KEY) and a portion of the central timing equipment (CTEO, CTCO, CTPA, CTSA, CTSB, CTOR) in conjunction with the HGA or the DSC. The central timing equipment subassemblies supply the carrier for the KEY. Voice reception is obtained in the same manner with the following exceptions: (1) the audio receiving amplifiers (ACRAA, ACRAB, ACRAC) are utilized, (2) the S-band power amplifier is not necessary for reception utilizing the DSC, and (3) the up-voice discriminator (PDVD) is used instead of the PDVR and PDVC.

The tracking and ranging aid is supplied by a portion of the S-band equipment (USBTA or its alternate, USBTB) in conjunction with either antenna.

TEST PROGRAM

Qualification Tests

SID 62-109 (General Test Plan Research and Development for Project Apollo), Volume III (Ground Qualification Tests) includes:

(1) Scope of test program

(2) System logic diagram

(3) Qualification test schedules

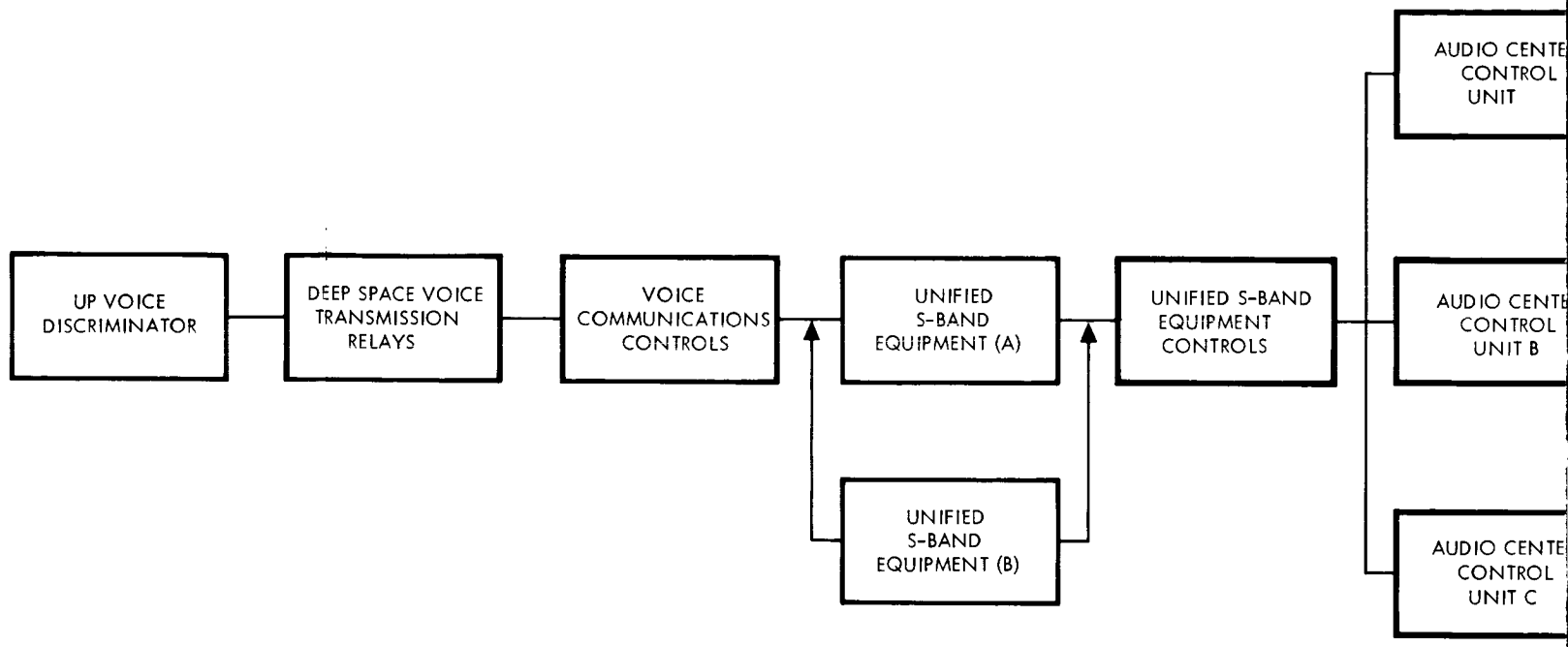
General Test Program Status

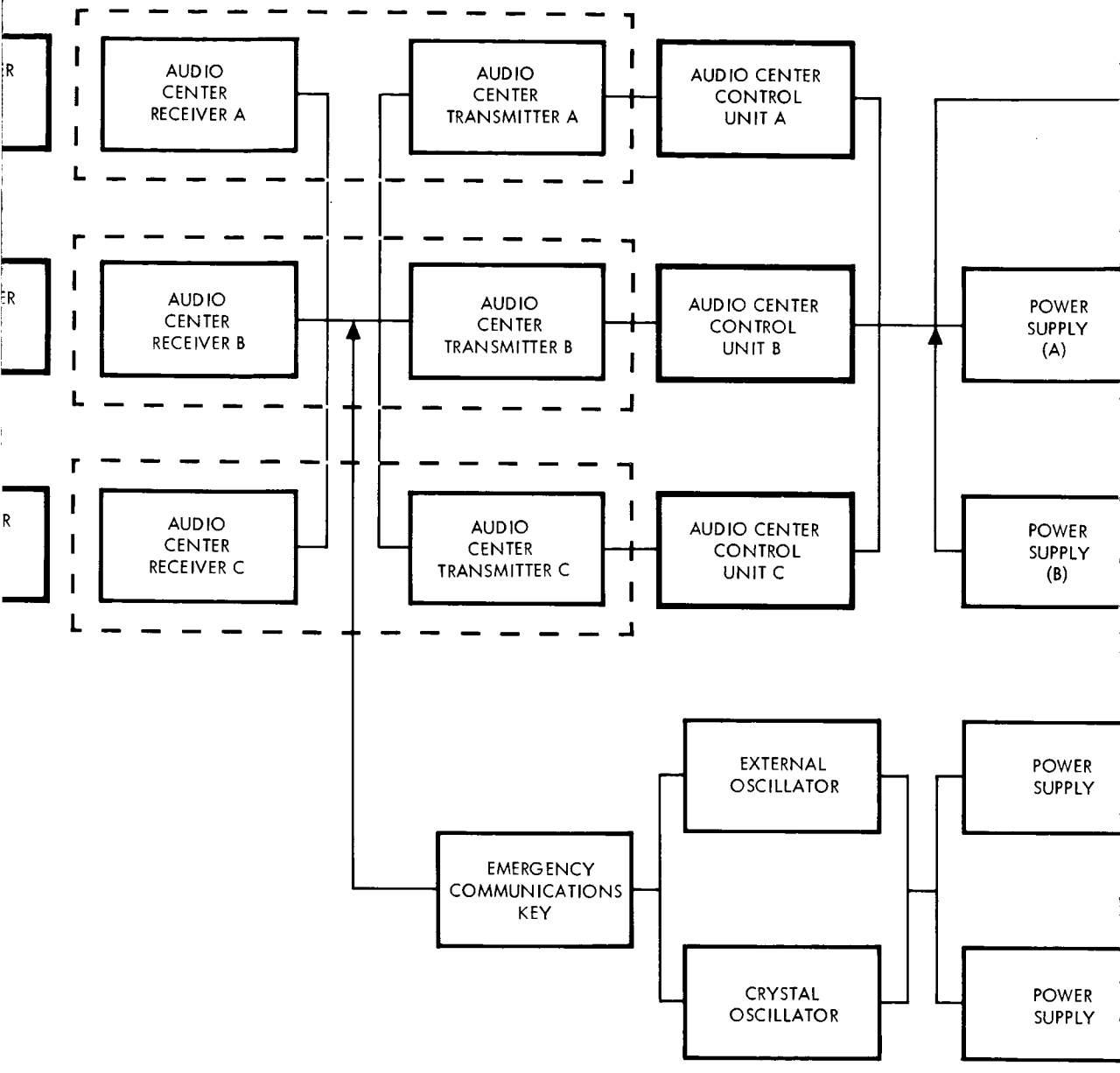
The development program is in the initial stages, with breadboards being constructed, parts being ordered, and certain price parts undergoing qualification tests.

It is anticipated that the first qualified C&D subsystem will be available in April 1964.

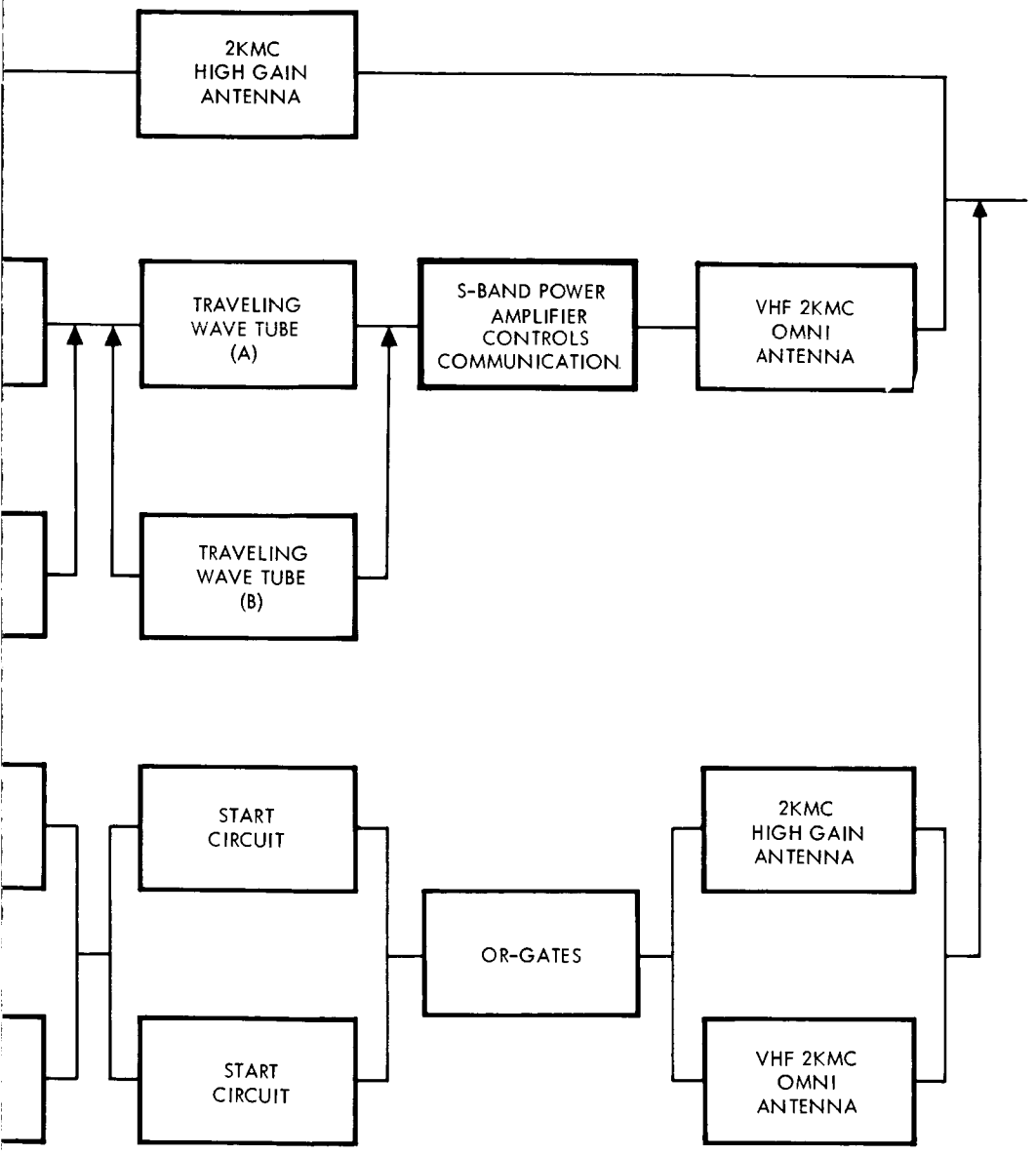
Specific Developmental Tests

Developmental tests have been conducted on four traveling-wave tubes at the Collins Radio Company. Failures resulted in all four of the tubes, the


UP VOICE DISCRIMINATOR
DEEP SPACE VOICE TRANSMISSION RELAYS
VOICE COMMUNICATIONS CONTROLS
UNIFIED S-BAND EQUIPMENT (A)
UNIFIED S-BAND EQUIPMENT (B)
UNIFIED S-BAND EQUIPMENT CONTROLS
AUDIO CENTE CONTROL UNIT
AUDIO CENTE CONTROL UNIT B
AUDIO CENTE CONTROL UNIT C


AUDIO CENTER RECEIVER A
AUDIO CENTER TRANSMITTER A
AUDIO CENTER CONTROL UNIT A
AUDIO CENTER RECEIVER B
AUDIO CENTER TRANSMITTER B
AUDIO CENTER CONTROL UNIT B
POWER SUPPLY (A)
AUDIO CENTER RECEIVER C
AUDIO CENTER TRANSMITTER C
AUDIO CENTER CONTROL UNIT C
POWER SUPPLY (B)
EXTERNAL OSCILLATOR
POWER SUPPLY
EMERGENCY COMMUNICATIONS KEY
CRYSTAL OSCILLATOR
POWER SUPPLY

Figure



4-22. Communications and Data Instrumentation Subsystem Subassemblies Logic Diagram

most significant cause of failure being a fault in the test set-up power supply protective circuit. The results of the failure analysis are given in SID 62-822-2, the Monthly Failure Report.

Exploratory-type induced environmental testing (vibration) is being conducted on the standard electronic packages (SEP's). A resonant search of the SEP's and modules, and investigations are being conducted of stress levels on through-bolts and holddown devices.

Breadboards are being constructed, and electrical functional tests will begin in the next reporting period.

Specific Qualification Tests

None performed during this report period. Qualification of piece parts and unique materials will be conducted during the next quarter.

SUBCONTRACTOR MANAGEMENT

NAA/S&ID Apollo Reliability Engineering performed an evaluation of the man-hours necessary for the subcontractor to perform the reliability effort required for the communications and data subsystem. This estimate was based on the ground rules and format supplied by the project managers office.

When the Collins Radio Company cost proposal was received, the S&ID Apollo Reliability estimate and the Collins cost proposal were compared, and differences in scope or effort were reconciled where possible. Unresolved points of difference were noted and discussed during the field analysis at the Collins Radio Company, held in February 1963.

Two problems hampered the comparison: (1) the lack of detailed information in some areas of the Collins proposal and (2) the difference in grouping of subitems under items and subtasks, even though there were few differences in the scope of work.

A reliability session was conducted during the field analysis, in which the general format of the cost proposal was followed. In lieu of covering the reliability effort of each equipment, the VHF/FM transmitter was used as an example of the Collins Radio in-house equipment, and the DSIF transponder was used as an example of the Collins subcontracted equipment. Equipment subtasks varying from the examples also were investigated. The results of this cost evaluation were presented to NASA-MSC from 5 through 8 March 1963 to establish a negotiation base for the communications and data subsystem contract. S&ID was requested to submit its

suggested base for negotiation of the reliability effort to NASA-MSC Reliability Engineering, which was subsequently accepted as realistic. The subcontract negotiations will begin in April 1963.

Reliability Failure-Rate Predictions

The preliminary reliability failure-rate predictions received from the subcontractor for the communications and data subsystem are shown in Tables 4-4 through 4-9.

Meetings

Representatives of S&ID Reliability attended a meeting at the Collins Radio Company in Dallas, Texas, to discuss the qualification test requirements for the Apollo R&D VHF-2KMC omni antenna and radome. Also present at the meeting were representatives from Goodyear Aircraft, supplier of the radome and foam material. Testing philosophy, levels and durations of tests, and test responsibilities also were part of the agenda.

The representatives of the Collins Radio Company, Dallas Division, and S&ID conducted a monthly negotiation meeting to determine the test environments for the R&D VHF/2KMC omni antenna, the R&D multiplexer, the quantity of units required for qualification, and changes in the responsibilities for testing. Facilities and construction techniques involved in certain phases of the test program necessitated that responsibility for tests be reassigned between Collins Radio and its associate subcontractors. During the meeting, Collins Radio presented to NAA the latest design and configuration techniques.

PLANNED ACTIVITIES

During the next quarter, the numerical requirements for the equipments of the communications and data subsystem will be revised to reflect the latest design changes. In conjunction with this effort, the subsystem sparing complement for the LOR mission will be refined. Also, the reliability logic diagrams for each equipment will be revised on both the sparable module and the part levels in a coordinated effort with the subcontractor.

Mandex worst case, parameter variation, and other computer analyses will be performed on circuits of the subsystem.

The failure-effects analyses at the subassembly level received from the subcontractor for the communications and data subsystem will be integrated with the failure-effects analyses for other subsystems for both AFRM 011 and the ultimate LOR mission.

The identification and traceability requirements will be defined for each part and equipment of the communications and data subsystem.

The Reliability Program Plan and other documentation for the subsystem will be reviewed, and comments forwarded to the Collins Radio Co.

Table 4-4. Unified S-Band Equipment (DSIF) Transponder - Subcontractor Predictions

Module	Failure Rate X 10^{-6} (hr)	MTBF (hr)
Receiver	24.27	41,100
Transmitter	12.16	82,200
Power supply	03.33	300,000
Total	39.76	*25,200
*Unified S-band equipment (DSIF) Transponder MTBF		

Table 4-5. Data Storage Equipment - Subcontractor Predictions

Module	Failure Rate x 10^{-6} (hr)	MTBF (hr)	No. of Modules	Failure Rate, All Modules	MTBF, All Modules
Digital reproduce amplifier	00.82	1,220,000	5	04.10	244,000
Analog record amplifier	00.87	1,150,000	9	07.83	128,000
Bias oscillator	00.22	4,550,000	1	00.22	4,550,000
Analog reproduce amplifier	01.48	675,000	9	13.32	75,000
Power supply	01.39	719,000	1	01.39	719,000
Digital record amplifier	00.42	2,380,000	5	02.10	476,000
Voltage regulator	01.28	781,000	1	01.28	781,000
Erase amplifier	01.18	847,000	1	01.18	847,000
Serial/parallel converter	03.38	296,000	1	03.38	296,000
Parallel/serial converter	03.38	296,000	1	03.38	296,000
System control logic	02.00	500,000	1	02.00	500,000
Tape transport	45.3	22,100	1	45.3	22,100
Total			36	85.48	11,700*

*Data Storage Equipment MTBF

Table 4-6. PCM Telemetry - Subcontractor Predictions

Module	Failure Rate s 10^{-6} (hr)	MTBF (hr)	No. of Modules	Failure Rate, All Modules	MTBF, All Modules
High-level analog gates	08.46	118,000	13	109.98	9,100
High-speed gates	06.05	165,000	1	06.05	165,000
A/D coder	30.53	32,000	1	30.53	32,800
Digital multiplexer	06.16	162,000	4	24.64	40,500
Clamped amplifier	13.66	73,200	1	13.66	73,200
Row and column drivers	19.27	51,900	2	38.54	26,000
Programmer counters	22.66	44,100	2	45.32	22,000
Programmer matrix	07.26	138,000	3	21.78	46,000
Programmer, miscellaneous	07.36	136,000	1	07.36	134,000
Output register	10.55	94,800	1	10.55	94,700
Low-level analog gates	07.18	139,000	5	35.90	27,800
Calibration generator	11.57	86,500	1	11.57	86,500
Power supply	09.36	107,000	1	09.36	107,000
Total			36	365.24	2,740*

*PCM Telemetry MTBF

Table 4-7. C-Band Transponder

A. The C-band transponder is not modularized.
B. The predicted failure rate is 12.88 percent/1000 hours.

Table 4-8. Pred Module Processor

Module	Failure Rate (%/1000 hr)	MTBF (hr)	Weight (oz)
1.25 MC VCO	0.116	861,000	10
(9) SCO's	0.451	222,000	40
Frequency multiplier	0.059	1,690,000	6
Module two			
Up voice discriminator	0.063	1,590,000	6
LP and BP filters	0.056	1,780,000	6
Mixing networks	0.015	6,670,000	1
Power supply	0.149	670,000	24
Chassis	0.060	1,670,000	64
Total	0.969	113,000	157

Table 4-9. Signal Conditioner - Subcontractor Predictions

Module	Failure Rate x 10^{-6} (hr)	MTBF (hr)	No. of Modules	Failure Rate, All Modules	MTBF, All Modules
D-c amplifier	02.60	384,000	22	57.20	17,500
D-c attenuator	03.20	312,000	4	12.80	78,000
A-c to d-c converter	01.68	595,000	6	10.08	99,200
Phase sensing demodulator	01.90	526,000	1	01.90	526,000
Frequency sensing demodulator	01.86	537,000	1	01.86	537,000
Power supply	04.40	227,000	1	04.40	227,000
Total			35	88.24*	11,300*

*Signal Conditioner MTBF

Table 4-10. Communications and Data Subsystem Failure Effects Analysis

Item	Drawing Number	Elect. Reference Desig.	Function	Failure Type	Mission Type	Failure Effect On Component Performance	Failure Effect On Subsystems Performance	Failure Effect On Apollo	Possible Crew Action
Premodulation processor			The premodulation processor equipment provides the interface connection between the spacecraft data gathering equipment and the r-f electronics, and performs switching, gating, and mixing associated with the various modes of information transmission.	Loss of subcarrier oscillators or associated mixing network	All phases		Loss of ability to process data storage output or real-time analog data for transmission	Loss of all digital sensory data	Limited transmission of data via voice and TV link and the visual displays
				Loss of up-voice discriminator			Loss of ability to process voice reception of the DSIF transponder	Loss of guidance and navigation backup	Abort
				Loss of low-pass filter			Loss of input to VHF/FM transmitter	Loss of primary mode of sensory data transmission in the near-earth phases	Limited transmission of data via voice and TV link and the visual displays
VHF/2 KMC omni antenna			The VHF/2 KMC omni antenna equipment provides the omni-directional radiating element and associated multiplexing and switching units for use with the DSIF and VHF equipments in all phases except earth landing and recovery.	Degradation of electrical properties	All phases except earth landing and post landing		Loss of VHF radiating element and secondary 2 KMC radiating element	Loss of primary voice and data transmission in near-earth phases and secondary voice and data transmission in deep-space phases	None

Table 4-10. Communications and Data Subsystem Failure Effects Analysis (Continued)

Item	Drawing Number	Elect. Reference Desig.	Function	Failure Type	Mission Phase	Failure Effect On Component Performance	Failure Effect On Subsystems Performance	Failure Effect On Apollo	Possible Crew Action
VHF recovery beacon			The VHF recovery beacon equipment provides line-of-sight-direction finding capabilities during the earth landing and recovery phases to aid in locating the spacecraft.	Short or open circuit output Oscillator drift	Earth landing and recovery phases		Failure to transmit Drift of carrier frequency	Loss of primary tracking during earth landing and recovery	Use HF transceiver keyed or VHF/AM CW beacon capability
HF transceiver			The HF transceiver equipment provides beyond line-of-sight direction finding and communication capabilities to aid in locating the spacecraft during the recovery phase.	Short or open circuit output Oscillator drift	Recovery phase		Failure to transmit Drift of carrier frequency	Loss of primary voice link during recovery phase	Use VHF/AM CW beacon and limited voice link of VHF/AM. (Line-of-sight transmission only)
VHF/AM transmitter-receiver			The VHF/AM transmitter-receiver equipment provides voice communications from the spacecraft to earth during the near-earth phases (up to 4000 miles) and extravehicular voice communications during the deep-space phases.	Loss of traveling wave tube Short or open circuit output Oscillator drift in transmitter or receiver section	Near-earth and circumlunar phases		Failure to transmit with full power Failure to transmit or receive Drift of carrier frequency in transmitter or receiver section	Loss of primary voice during near-earth and circumlunar phases	Transmit at reduced power. Use of DSIF equipment in conjunction with VHF/2 KMC omni antenna Transmit at reduced power. Use of DSIF equipment in conjunction with VHF/2 KMC omni antenna Transmit at reduced power. Use of DSIF equipment in conjunction with VHF/2 KMC omni antenna

Table 4-10. Communications and Data Subsystem Failure Effects Analysis (Continued)

Item	Drawing Number	Elect. Reference Desig.	Function	Failure Type	Mission Phase	Failure Effect On Component Performance	Failure Effect On Subsystems Performance	Failure Effect On Apollo	Possible Crew Action
VHF/FM transmitter			The VHF/FM transmitter equipment provides transmission of telemetry information during the near-earth phases (up to 4000 miles)	Oscillator drift Tube failure Short or open circuit output	Near-earth phases		Drift of carrier frequency Failure to transmit Failure to transmit	Partial loss of telemetry data; total loss of real-time undisplayed data	Use of data storage equipment for limited time. Use of VHF/AM transmitter receiver and visual displays (limited data)
Audio center			The audio center equipment provides amplification of audio voice transmission and reception, and automatic voice-operated transmission (VOX) for each crew member in all phases.	Loss of receiving amplifier Loss of transmitting amplifier Loss of VOX control	All phases		One failure, no effect; two failures, loss of one audio link; three failures, loss of two audio links; four failures, loss of function	Two to three failures, degradation of voice communications; four failures, loss of voice communications	One failure, replace with spare; two to three failures, no action; four failures, use key (in conjunction with central timing equipment in deep-space phases)
Data storage			The data storage equipment provides on-board telemetry data storage during critical periods and when real time data transmission is not possible and provides playback for delayed time data transmission.	Tape breakage Mechanical failure	Powered phases Earth entry and far side of moon		Temporary loss of data storage and playback capabilities Loss of data storage and playback capabilities	Temporary loss of data storage and playback capabilities Loss of data storage and playback capabilities	Voice relay of delayed time data from visual displays (limited data)

Table 4-10. Communications and Data Subsystem Failure Effects Analysis (Continued)

Item	Drawing Number	Elect. Reference. Desig.	Function	Failure Type	Mission Phase	Failure Effect On Component Performance	Failure Effect On Sub-Systems Performance	Failure Effect On Apollo	Possible Crew Action
Data storage (Continued)				Short or open circuit amplifier output			Loss of data storage and playback capabilities	Loss of data storage and playback capabilities	
C-band transponder			The C-band transponder equipment provides response to interrogations from complementary ground radars during the near-earth phases (up to 4000 miles) for the purpose of enhancing and extending the ground tracking capability.	Tube failure Oscillator drift Short or open circuit output	Near-earth phases		Loss of response to interrogations of ground radar for tracking and ranging	Loss of secondary mode for G&N function	Use DSIF in conjunction with VHF/2 KMC omni antenna when within range of DSIF ground stations
DSIF transponder			The DSIF transponder equipment provides voice communications and transmission of telemetry data, TV, and tracking information from the spacecraft to earth during the deep-space phases (over 4000 miles).	VCO failure Ranging code generator inoperative	All deep-space phases		Loss of phase coherence between CM and GOSS causing loss of function Loss of ranging capability	One failure, no effect; two failures, loss of secondary mode for G&N function Loss of ranging capability	One failure, use spare transponder; two failures, abort. Emergency voice available via crystal oscillator Voice backup with visual displays information

Table 4-10. Communications and Data Subsystem Failure Effects Analysis (Continued)

Item	Drawing Number	Elect. Reference. Desig.	Function	Failure Type	Mission Phase	Failure Effect On Component Performance	Failure Effect On Subsystems Performance	Failure Effect On Apollo	Possible Crew Action
DSIF power amplifier			The DSIF power amplifier provides power for the primary transmission mode of the DSIF equipment.	Tube failure Short or open circuit output	All deep space phases		Loss of wide-band telemetry TV and ranging capabilities. Degradation of voice capability	One failure, no effect; two failures, loss of function	One failure, use spare power amplifier; two failures, use DSIF transponder with high-gain antenna for narrow-band transmission
Signal conditioner			The signal conditioning equipment provides normalization for all sensory data prior to input to PCM telemetry equipment.	Open circuit output Short circuit Degradation of output	All phases		Loss of data channel Loss of data channel Erroneous data transmission	Loss of data from affected channel Loss of data from affected channel Erroneous data from affected channel	None
PCM telemetry			The PCM telemetry equipment pulse code modulates and serializes all sensory data for all phases.	Failure of analog to digital converter Failure of master digital programmer	All phases		Loss of ability to digitize sensory data	Loss of all digital sensory data	Limited transmission of data via voice and TV link and the visual displays

GUIDANCE AND NAVIGATION

SUMMARY

Studies related to the guidance and navigation subsystem presently under way at NAA/S&ID are designed as follows:

1. To specify the requirements for the guidance and navigation system--spacecraft interfaces and over-all performance and reliability-- in detail

2. To select a spacecraft configuration and an allotment of on-board spares that will assure the attainment of Apollo Spacecraft reliability objectives

3. To conduct continuous detailed quantitative system analyses to assess the adequacy of the systems to meet mission success and crew safety objectives

The objective of these studies is to emphasize the relationship between reliability and mission operations. Particular attention is being directed toward alternate modes of operation and their impact on system reliability and backup requirements. Evaluations are being conducted to optimize the character, as well as the number, of on-board spares.

ANALYSIS

Analyses performed by NAA/S&ID to determine the predicted reliability of the guidance and navigation subsystem, based on Apollo preferred or equivalent high reliability parts compared with Polaris parts, yielded the results shown in Table 4-11.

Apollo Guidance and Navigation System Reliability Apportionments and Initial Analysis, MIT Report R-395, included a prediction of guidance and navigation subsystem reliability based on present state-of-the-art parts and "design excellence." The results of this study are shown in Table 4-12. These data have been included in the mission simulation analysis.

NAA/S&ID and MIT conducted their studies independently. The studies indicated that the analytical methods used by NAA/S&ID and MIT are consistent. As determined by MIT, the prediction of the basic reliability of

Table 4-11. NAA/S&ID Guidance and Navigational Subsystem Comparative Apportionment and Predictions

Equipment	Mission Time Hours	Mission Success Reliability		
		Apportioned (NAA/S&ID)	Predicted* (No Redundancy or Spares)	
			Apollo Preferred Parts	Polaris Parts
Inertial measurement unit	39.88	0.99720	0.967766	0.964701
Power and servo assembly	39.88	0.99990	0.998532	0.991035
Coupling display unit	39.88	0.99900	0.995119	0.992739
Apollo guidance computer	66.07	0.99900	0.997282	0.932605
Space sextant	55.02	0.99900	0.994469	0.994469
Scanning telescope	55.02	0.99990	0.999362	0.999362
Totals		0.9940125	0.953132	0.884285

*Based on State-of-the-Art failure rates

Table 4-12. MIT Guidance and Navigation Subsystem Apportionment and Predictions

System	Operating Time (hr)			Mission Success Reliability	
	Full Power	Standby	Off	Apportioned (MIT)	Predicted (Without Spares)*
Coupling display unit	22		135	0.999983	0.99652
Sextant and scanning telescope	22		135	0.998585	0.99787
Apollo guidance computer	45	112		0.998667	0.89826
Inertial measurement unit gimbal assembly	22		135	0.998114	0.99626
Inertial measurement unit electronic	22	135		0.999333	0.9932
Space sextant and scanning telescope	22		135	0.999334	0.99635
Final approach equipment	2		155	0.999983	0.99992
Total				0.994	0.88054

*Based on State-of-the-Art failure rates.

the guidance and navigation system is 0.88054 (Table 4-12). This prediction, determined by NAA/S&ID with comparable parts, is 0.88485. On the other hand, a prediction of the basic reliability determined by NAA/S&ID and based on the use of Apollo preferred or equivalent high reliability parts was 0.953132 (Table 4-13). While MIT appears willing to accept the 0.88054 value with the anticipation of using maintenance actions and spares to reach the 0.994 apportioned value, NAA/S&ID favors the 0.953132 value because it would serve to minimize in-flight maintenance and spares replacement. The difference between NAA/S&ID and MIT, therefore, is not one of technical approach but rather one of reliability philosophy.

Table 4-13. Inertial Measurement Unit Failure Rate Comparisons

Equipment	MIT Listed Failure Rate X 10^{-6} (hr)	* S&ID Parts Counts Failure Rate X 10^{-6} (hr)	MIT S&ID Difference	Apollo High Reliability Failure Rates X 10^{-6} (hr)
PSA6 Current Switch	7.701	7.385	-0.316	1.3788
PSA6 DC Differential Amplifier	2.756	2.775	+0.019	1.344
PSA3 AC Differential Amplifier	3.811	3.365	-0.446	0.734
PSA3 Interrogator	5.516	5.440	-0.076	0.848
PSA3 PIP Calibration Module	1.616	1.80	+0.184	0.800
PSA3 Gyro Calibration Module	1.716	1.70	-0.016	0.760
PSA3 Gimbal Servo Amplifier	2.671	2.625	-0.046	0.4622
PSA3 Gimbal Course Align	3.191	4.250	+1.059	3.009
PSA3 -28-v d-c P/S	3.82	3.545	-0.275	0.4761
Total	32.798	32.885		9.8121

*Using component failure rates from MIT listed in Table 4, R-395

The attainment of the reliability objective for mission success and crew safety requires high reliability parts in mission-essential equipments of the spacecraft. The importance of this requirement takes precedence over the function of in-flight maintenance in the assurance of equipment availability throughout various mission phases.

Module-Level Configuration Analysis

The analysis during this period consisted of breaking down equipments of the guidance and navigation system to the module level. The equipments used in the process of making mid-course navigational sightings are shown in Figure 4-23. The logic diagram is preliminary and will be updated as additional information is received from MIT.

Failure Effects Analysis

The results of a preliminary failure effects analysis of the guidance and navigation subsystem at the equipment level are given in Table 4-14 at the end of this section.

Inertial Measurement Unit Failure Rate Comparisons

Comparisons have been made between the failure rate predictions for nine IMU electronic circuits (for which circuit diagrams have been received from MIT) and predictions based upon Apollo preferred high reliability parts. The results of the study are shown in Table 4-13. Included in the table is a list of circuits with their corresponding failure rates taken from MIT Report R-395. Failure rates calculated by NAA/S&ID based upon parts counts and individual part failure rates (MIT Report R-395) are shown. Since the differences between the MIT failure rates and the S&ID calculated failure rates indicate a +33 to -12 percent range from MIT numerical values, more than methodology is apparently involved. It seems imperative that information be made available, so that justified accounting can be given for deviations that may be experienced in the future. Failure rate predictions anticipated with Apollo preferred high reliability parts are listed. Results show a significant difference in reliability representing approximately one order of magnitude.

SUBSYSTEM DESCRIPTION (G&N)

Inertial Measurement Unit

The inertial measurement unit (IMU) is a three-axis gimbal assembly designed to space stabilize the stable member (SM). The SM is supported by the middle gimbal (MG) and free to rotate with respect to the MG about

the inner axis. Similarly, the MG is supported by the outer gimbal (OG) and rotates about the middle axis, and the OG is supported by the gimbal case (GC) and rotates about the outer axis.

The SM supports six inertial components (IC) suitably oriented to each other; these are three size 25 inertial reference integrating gyros (25 IRIG) and three size 16 pulsed integrating pendulums (16 PIP). Also mounted on the SM are the necessary gimbal-mounted electronics (GME) modules associated with these inertial components.

The SM, MG, and OG are each supported, respectively, in the MG, OG, and GC by a pair inter-gimbal assemblies (IGA). The inter-gimbal assemblies consist of a stub shaft, bearings, bearing support, and such items as torquers, electromagnetic components, and slip ring assemblies, as are required for the axis in question.

The axis orientation of the IMU is such that with all gimbal angles zero (i. e. , gimbal axis orthogonal), the inner axis will be aligned with the spacecraft pitch axis, the middle axis with the spacecraft yaw axis, and the outer gimbal axis with the spacecraft roll axis. All axes will have unlimited angular freedom. To avoid gimbal lock, provision will be made for accurately reorienting the stable member about the middle axis. The IMU will be space aligned by means of space sextant information, and because of this, the IMU and space sextant will be mounted in close proximity, with as much rigidity as necessary.

The inertial reference integrating gyroscopes have an angular momentum of 450, 000 gm-cm^2 sec. These are floated integrating gyroscopes and geometrically stabilized in the case by the fluid and the magnetic suspension system (ducosyn). Signal information is transmitted by a microsyn signal generator. A microsyn torque generator will be utilized for realigning the IMU in flight and for preflight erection and alignment. The gyroscope will be pulse torqued for these alignments.

The accelerometers are pulse-torqued pendulums using the 16 PIP, as designed for the Apollo mission, and provide velocity increments of the integrated acceleration. The pendulums are geometrically stabilized in the case by the flotation fluid and the magnetic suspension (ducosyn). The pendulum also contains a microsyn signal generator and torque generator.

The IMU will provide a space stabilized coordinate system and measure the specific force. It may be used in the initial earth-launch guidance system. It will be used for rendezvous guidance, orientation and guidance for injection into the earth-lunar orbit, mid-course velocity correction measurements, guidance for lunar landing and lunar takeoff and guidance for

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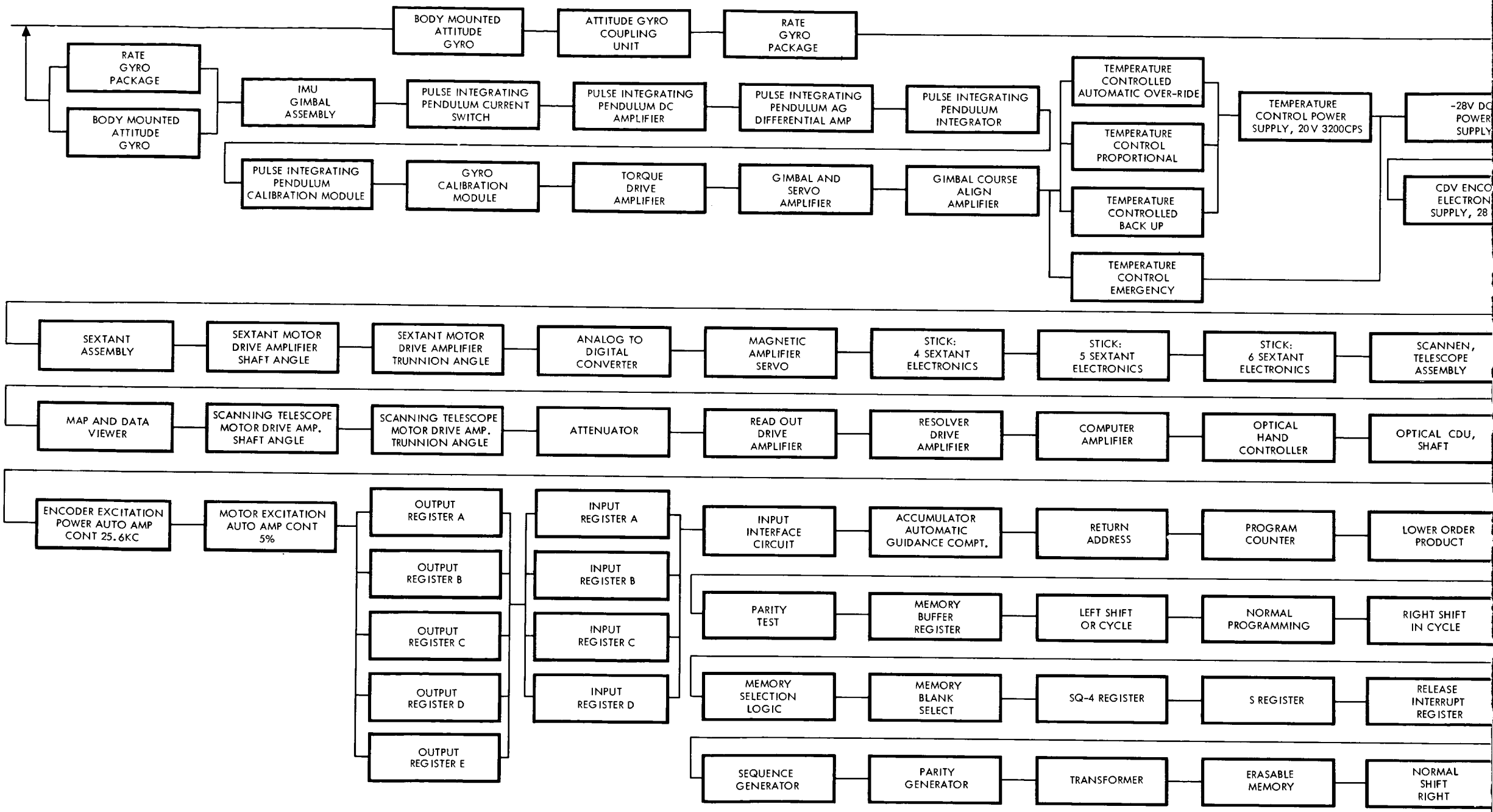

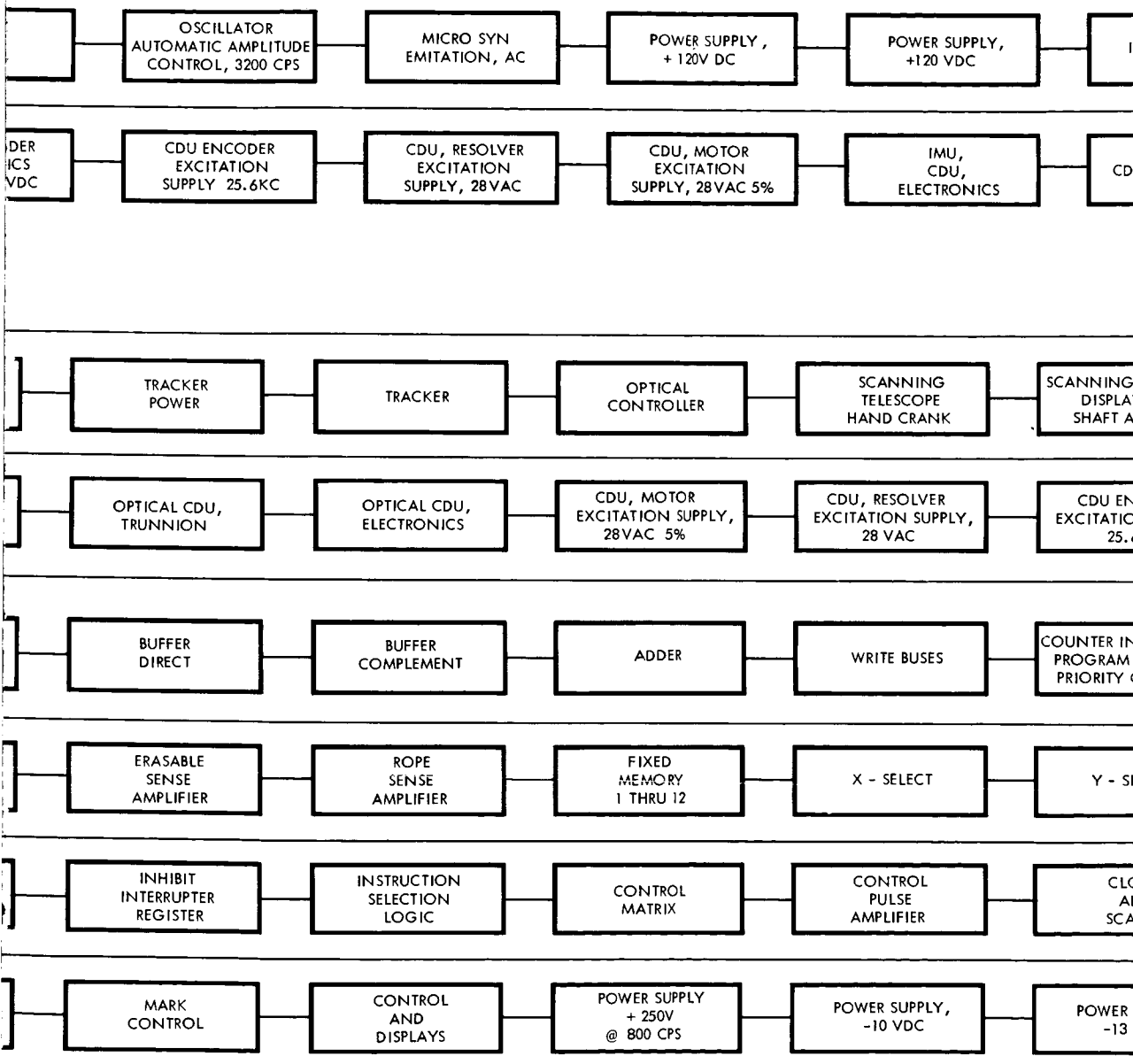
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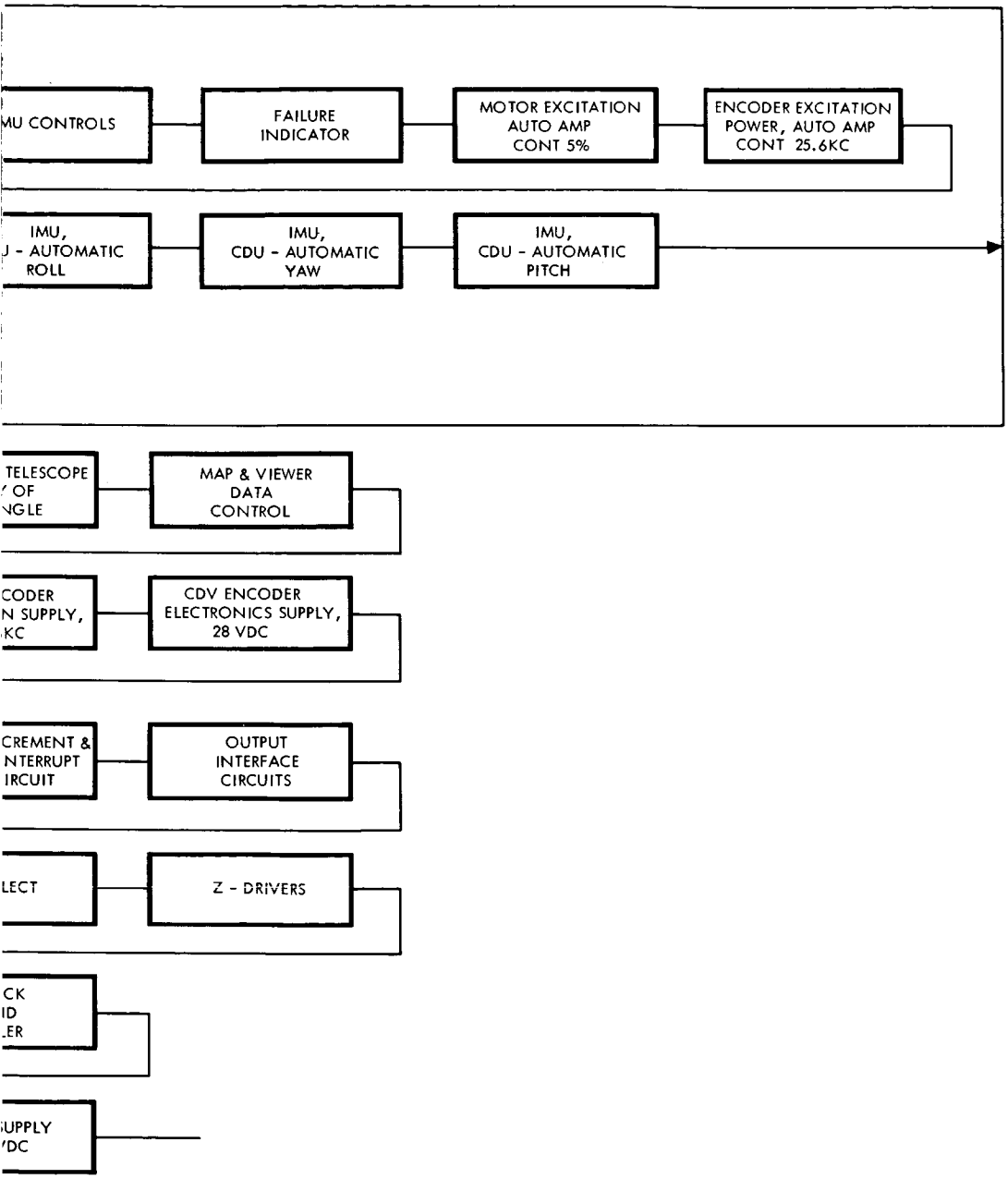
MICRO SYN EMITATION, AC
POWER SUPPLY, + 120V DC
POWER SUPPLY, +120 VDC
CDU ENCODER EXCITATION SUPPLY 25.6KC
CDU, RESOLVER EXCITATION SUPPLY, 28VAC
CDU, MOTOR EXCITATION SUPPLY, 28VAC 5%
IMU, CDU, ELECTRONICS
TRACKER POWER
TRACKER
OPTICAL CONTROLLER
SCANNING TELESCOPE HAND CRANK
OPTICAL CDU, TRUNNION
OPTICAL CDU, ELECTRONICS
CDU, MOTOR EXCITATION SUPPLY, 28VAC 5%
CDU, RESOLVER EXCITATION SUPPLY, 28 VAC
BUFFER DIRECT
BUFFER COMPLEMENT
ADDER
WRITE BUSES
ERASABLE SENSE AMPLIFIER
ROPE SENSE AMPLIFIER
FIXED MEMORY 1 THRU 12
X - SELECT
INHIBIT INTERRUPTER REGISTER
INSTRUCTION SELECTION LOGIC
CONTROL MATRIX
CONTROL PULSE AMPLIFIER
MARK CONTROL
CONTROL AND DISPLAYS
POWER SUPPLY + 250V @ 800 CPS
POWER SUPPLY, -10 VDC

Figu



re 4-23. Guidance and Navigation System Logic Diagram

the earth reentry. Temperature control of the inertial components and heat dissipation for all power consuming items, such as the inertial components, torquers, and electromagnetic components, will be designed into the IMU.

Space Sextant

The space sextant will be primarily a visual instrument to be used by the astronaut to:

1. Measure the angle between line of sight to a star and line of sight to a landmark on the earth or lunar surface

2. Measure the elevation of line of sight to a star above horizon of the earth or moon

3. Detect the occultation of a star by the earth or moon

4. Provide erection signals for the inertial measurement unit

The instrument, as presently conceived, will consist of an optical assembly and a controller assembly. The optical assembly will have two lines of sight—incorporation high magnification optics and incorporation low magnification optics with a wider field. Means for precise reading of the angle between two selected lines of sight will be provided. Articulation of the lines of sight will be accomplished by rotating the sextant head about an axis (shaft axis) perpendicular to the spacecraft surface and a second axis (trunnion axis) perpendicular to the lines of sight and the shaft axis. Depending on the results of research in high vacuum environment, the instrument will be mounted in the space environment (preferred) or behind an optically flat window.

Some spacecraft attitude maneuvering will be required to make sextant readings. The sextant controller will be used by the astronaut to provide resolved drive signals for the sextant and spacecraft attitude stabilization system during readings. The controller will also read the sextant angles into the Apollo guidance computer.

Scanning Telescope (SCT)

This instrument is essentially an articulated telescope with two degrees of freedom and a variable focal-length eyepiece, which adjusts the magnification and field continuously between 1 and 3 power and 60 and 20 degrees, respectively. The SCT is equipped with both manual and servo drives. Electrical gear-tooth counter encoders and mechanical angle counters are provided for automatic and visual angle readouts, respectively.

The command module SCT is used for acquisition and identification of celestial bodies and for tracking of landmarks that have high angular rates with respect to the spacecraft. It is also used as a backup to the space sextant for star tracking during alignment of the IMU.

Computer

The principal structure of the computer consists of central flip-flop registers, the majority of which are addressable. The central registers communicate with one another and with the rest of the computer via write buses. Gating pulses for reading out of, and writing into, the central registers are formed in the control pulse amplifiers. For addressable registers, gating pulses are dependent, in part, upon memory selection logic.

Memory addressing is effected via registers that are used for memory bank selection. The oscillator output for the computer is divided into sub-harmonics with intermediate frequencies being used to perform specific computer functions. A controlled signal is taken from the computer oscillator output to serve as a synchronizing signal to the central timing equipment for the spacecraft.

The sequence generator of the computer directs time pulses, so that selected instructions will be executed after all counter requests are fulfilled. Other functions are under the central processing section, memory operation, parity test and generation, counter incrementing and wait list, and executive programs. Program interruption is provided by release-interrupt or inhibit-interrupt signals.

Computer programs are prepared to interpret pseudo-codes or interpretive instructions. These pseudo-codes include a variety of double-precision operations and a small number of triple-precision operations, together with a set of double-precision vector operations.

The computer functions to supply IMU pick-off information, accelerometer readouts as increments of velocity, sextant angle and drive readouts, and information for pilot display. It also serves as the basic unit for the display of information communicated to the spacecraft via the up data link. In addition to the computer function of computing projected trajectories for comparison with specific trajectories calculated for the normal spacecraft flight path, it is the primary source for determining the precision roll commands necessary to assure survival during entry maneuvers.

Coupling Display Unit

The coupling display units are servo components that enable the computer to communicate with the inertial measurement unit, the optics,

and the stabilization and control subsystem. The coupling display unit is composed of shafts driven by torque motors that may be controlled through a digital-to-analog converter by the computer. Incremental encoders mounted on the shafts supply the computer with information concerning the position of the coupling display unit shaft relative to IMU gimbal or optical shaft angle positions. This information is acquired from error signals, measured as differences in outputs, as determined by coupling display unit, inertial measurement unit, and optic component positions.

Power and Servo Assembly

Electrical apparatus for the distribution of power and signal conditioning within the G&N subsystem is contained in the power and servo assembly. The ternary current switch provides torquing signals to the pulse integrating pendulum accelerometers (PIPA) and to the inertial rate integrating gyros (IRIG), as well as ΔV signals to the computer. The d-c differential amplifier accepts demodulated coupling display unit (CDU) resolver signals and feeds them to the torque drive amplifier during the course align mode. Angular differientating accelerometer signals are also accepted from the IMU and fed into the torque drive amplifier. The a-c differential amplifier amplifies the coupling display unit one-time resolver signal for the 800-cycle demodulator. The interrogator reshapes the output wave of the gyro pick-off analog output to form an approximate square wave, and it detects the phase and magnitude of the wave form.

The pulse integrating pendulum calibration module provides a torquing feedback signal from the pulse torquing electronic circuits to the pulse integrating pendulum accelerometer. The gyro calibration module provides a feedback signal from the gyro pulse torquing electronics to the inertial rate integrating gyro assembly. The torque drive, gimbal servo amplifier provides excitation for the inertial measurement unit gimbal motors. The motor drive amplifier provides power to the coupling display unit shaft drive motors. The gimbal course align amplifier provides alignment signals to the inertial measurement unit gimbal torque motors in the course align mode. Power supplies include the 800-cps, 28-volts, 1-percent power amplifier, the 800-cps, 28-volts, 5-percent power amplifier, the 25.6-kc 28-volts, 1-percent power supply, and the 3200-cps, 20-volts, square wave, temperature control power supply. The 28-volts, d-c power serves to supply power for the temperature controller in the emergency mode.

PROBLEM AREAS

Considerable difficulty has been experienced in NAA/S&ID's endeavor to establish mission operating times of guidance and navigation subsystem equipment for computation of reliability predictions. IMU Preliminary

Operational Duty Cycle, MIT Report E-1092, is based upon a 190.5-hour mission time. The total mission operating time for the IMU indicated by this report is 36.93 hours. The report concluded with IMU worst-case duty cycles of 12.25 hours for earth departure maneuvers, and 5.60 hours for lunar maneuvers. Based upon a 157-hour mission profile time, the derivation of reliability values found in Apollo Guidance and Navigation System Reliability Apportionments and Initial Analysis, MIT Report R-395, shows a 22-hour operational time for the IMU gimbal assembly. Two subsequent reports have been received from MIT. "Apollo Mission Phase Sequences" that is based upon a total mission time of 139 hours, and "Equipment Operation Modes" in which 31.35 hours are reported as the time if IMU operation. NAA/S&ID Apollo Reliability Engineering has submitted to MIT a 14-day maximum mission profile time line showing a 64.972-hour IMU operating time and a 194-hour nominal mission profile time line, in which the IMU operating time is reported as 55.42 hours.

The basic differences requiring resolution are equipment mission operating times for reliability computations. Use was made of times of 5 hours for the computation of the reliability values reported in MIT Project Memo 116, 22 hours in MIT Report R-395, and 31.35 hours in Equipment Operating Modes. NAA/S&ID Apollo Reliability Engineering used a value of 39.88 hours as the IMU operating time in the reliability mission simulation studies.

NAA/S&ID Apollo Reliability Engineering current mission profile time lines indicate higher values of 64.97 hours maximum and 55.42 hours minimum. The IMU gimbal assembly operational time was used to illustrate the point. For each of the equipment used in the guidance and navigation subsystem, the need for establishing, by coordination between MIT and S&ID, definite operating times for reliability analyses is apparent.

ASSOCIATE CONTRACTOR COORDINATION

Primary associate contractor coordination has been through coordination meetings and coordination documentation. The seventh coordination meeting of NASA/MT/NAA-S&ID was held at NAA/S&ID on 15 and 16 January 1963. To resolve the problem of NAA/S&ID acquisition of data from MIT, a special technical meeting on reliability will be held at MIT between NAA/S&ID and MIT prior to a regular coordination meeting. This meeting has been delayed until the next quarter at the request of MIT.

PLANNED ACTIVITIES

Concerted reliability efforts will be made to improve and sustain coordination among NAA/S&ID, MIT, and Grumman. NAA/S&ID will

investigate assembly failure effects on the performance of the components, subsystems, and spacecraft. Requirements will be more firmly established for redundancy, spares, maintenance, and/or crew action. Efforts will continue toward the acquisition of detailed parts information needed to support the NAA/S&ID evaluation of the reliability of the guidance and navigation subsystem and the LEM. Such evaluations will be made to assure compatibility of the guidance and navigation subsystem in performance and reliability interfaces at the module, assembly, subsystem, and spacecraft levels.

Lunar excursion module efforts will center on the anticipated sequence of developmental steps necessary to establish the configuration, definition, philosophy, performance, and reliability interface relationships between Apollo and LEM. To implement the necessary interface technical coordination, NAA/S&ID Apollo Reliability Engineering will request current information on related Apollo/LEM systems and assemblies from the LEM contractor.

Analyses will be undertaken to establish the logic of the functional performance and reliability relationships, alternate modes, and backup procedures in the communications and data, guidance and navigation, stabilization and control, and instrumentation subsystems. Special studies and failure effects analyses to establish the framework for the integration of the guidance and navigation subsystems of the LEM and the spacecraft and related systems will be undertaken.

Table 4-14. Guidance and Navigation Failure Effects Analysis

Item	Drawing Number	Elect. Reference. Desig.	Function	Failure Type	Mission Phase	Failure Effect On Component Performance	Failure Effect On Sub-Systems Performance	Failure Effect On Apollo	Possible Crew Action
IMU			Attitude and attitude error signal to electronics display	Size 16 IRIG error; middle gimbal error; outer gimbal error; X, Y, Z PIP error; IMU + 28-volt power loss; 3200-cps loss; wheel 800-cps loss; resolver tachometer 800-cps loss; 28-volt d-c loss		Erroneous or no outputs	1-Incorrect or no attitude reference causing possible drift. 2-Incorrect orientation on thrust causing larger terminal errors.	Abort	Switch to SCS mode.
IMU			Delta velocity signal to AGC	Size 16 IRIG error; middle gimbal error; outer gimbal error; X, Y, Z PIP error; IMU + 28-volt power loss; 3200-cps loss; wheel 800-cps loss; resolution tachometer 800-cps loss; 28-volt d-c loss		Erroneous or no outputs	Engine will not shut off properly during maneuver. If display or indicator is too fast, engine will be manually shut down too early. If indicator is too slow and automatic system does not function properly an over-connection will be made and waste fuel.	Abort	Use ΔV display ON-OFF controls and try to repair if possible.
SXT hand controller			Precision roll command signal to AGC	SXT LM trunnion angle; SXT shaft angle; SXT SL trunnion angle		Erroneous or no outputs	No N&G navigational sighting capability.	Possible abort or loss of spacecraft.	Use SCS mode and try to repair.
AGC			Precision roll command signal to SCS electronics	Computer power loss; computer error		Erroneous or no outputs	No N&G navigational sighting capability.	Possible abort or loss of spacecraft	Use SCS mode and repair if possible.
AGC			Command attitude signal to CDU's	Computer power loss; computer error		Erroneous or no outputs	No N&G attitude control.	Abort	Use SCS mode and repair if possible.

Table 4-14. Guidance and Navigation Failure Effects Analysis (Cont)

Item	Drawing Number	Elect. Reference. Desig.	Function	Failure Type	Mission Phase	Failure Effect On Component Performance	Failure Effect On Sub-Systems Performance	Failure Effect On Apollo	Possible Crew Action
AGC			Attitude signal to attitude indicator	Computer power loss; computer error		Erroneous or no outputs	None	None	Repair.
CDU			Attitude error signal to error indicator	Motor excitation loss; inner CDU error; middle CDU error; outer CDU error		Erroneous or no outputs	If signal to autopilot is functioning correctly, the vehicle will maintain attitude but no error will be indicated.	None	Repair if possible.
CDU			Attitude error signal to error indication and SCS electronics	Motor excitation loss; inner CDU error; middle CDU error; outer CDU error		Erroneous or no outputs		If repair is impossible because of extended use required of the SCS system, a large degradation of reliability would result; possible abort.	Repair if possible after checking by use of external reference.
CDU			Attitude error signal to SCS electronics	No signal to autopilot		Erroneous or no outputs	Failure to change or hold correct attitude.	Loose accuracy; possible abort.	After switching to SCS mode, repair if possible.
AGC			Thrust ON-OFF signal to SCS electronic engine control	Computer error; computer power failure		Erroneous or no outputs	Engine will not react to commands.	If repair cannot be done, degradation of reliability of SCS will happen due to excess operating time required; possible abort.	After switching to SCS mode, repair if possible.

Table 4-14. Guidance and Navigation Failure Effects Analysis (Cont)

Item	Drawing Number	Elect. Reference. Desig.	Function	Failure Type	Mission Phase	Failure Effect On Component Performance	Failure Effect On Sub-Systems Performance	Failure Effect On Apollo	Possible Crew Action
AGC			Time-to-go to thrust signal to time-to-go display	Computer power loss; computer error		Erroneous or no outputs	None as long as automatic system is functioning properly.	None	Check and repair.
SCT			Trunnion angle shaft angle signal to AGC	Shaft zero		Erroneous or no outputs	Degradation of performance.	Limited sighting accuracy	None if it cannot be repaired.
AGC power supply dc			-10 volts (rms), -13.5 volts (rms) and +1-3.0 volts (rms) signals to AGC	Power loss		Erroneous or no output	Failure of automatic N&G mode.	If it cannot be replaced, abort.	Replace failed stick in PSA.
PSA electronics d-c minus supply			-28-volt signal to IMU/servo amplifier	Power loss		Erroneous or no outputs	Complete failure of IMU and possibly N&G system.	Possible abort.	Repair if possible.
PSA 800-cps source			28-volt (rms) signal to gyro wheel blower (IMU) resolver tachometer servo motor excitation	Power loss		Erroneous of no output	Loss of N&G automatic mode	Abort if it cannot be repaired.	Repair if possible.
PSA 3200-cps source			2-volt signal to gyro PIP's	Power loss		Erroneous or no output	Failure of N&G for maneuver.	None if stick is replaced; if it cannot be replaced, abort.	Replace stick in PSA.
PSA 25.6 kc			5-volt signal to sextant shaft to digital	Power loss		Erroneous or no output	No N&G navigation sighting capability.	None	Repair if possible by replacing stick in PSA.
PSA 25.6 kc			5-volt signal to IMU to CDU	Power loss		Erroneous or no output	Loss of N&G automatic mode.	None if it can be replaced.	Repair if possible with replacing stick in PSA.

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STABILIZATION AND CONTROL

SUMMARY

Equipment of the stabilization and control subsystem are constantly being subjected to reliability analyses to ensure that each component attains its optimum reliability. Because of the complexity of the subsystem, some problems have arisen. Such studies as failure effects analyses, circuit analyses, and sparing versus operational redundancy have been performed to realize solutions for these problem areas.

ANALYSIS

Analyses have been performed by Minneapolis-Honeywell, which indicate that the mission success reliability apportionment can be met with a reasonable complement of on-board spares.

The preliminary subassemblies for this equipment have been determined, and configuration breakdowns are shown in Figures 4-24 through 4-45. A comparison of failure rates and weights of each subassembly being used by NAA/S&ID and Minneapolis-Honeywell are shown in Table 4-15. The logic used in this configuration is oriented toward maximum compatibility with the equipment mission-phase logic diagrams for safe abort. These diagrams appear at the end of the introductory portion of the Electronic Systems Analysis Section.

Table 4-15. Comparison of NAA/S&ID and Minneapolis-Honeywell Failure Rates

Subassembly	Quantity	Failure Rates X 10^{-6} (hr)		Weight (lb) Each
		NAA	M-H	
Accelerometer	1	125.0	50.0	0.62
Accelerometer amplifier demodulator	1	2.49	4.317	0.75
Accelerometer integrator	1	4.5		1.5

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Table 4-15. Comparison of NAA/S&ID and Minneapolis-Honeywell Failure Rates (Continued)

Subassembly	Quantity	Failure Rates X 10^{-6} (hr)		Weight (lb) Each
		NAA	M-H	
Accelerometer mode select power supply	1	17.052	2.811	1.5
Accelerometer TC amplifier oscillator	1	2.746	1.989	0.75
Roll body gyro	1	150.0	200.0	1.2
Pitch body gyro	1	150.0	200.0	1.2
Yaw body gyro	1	150.0	200.0	1.2
Roll Preamplifier buffer amplifier	1	2.353	1.367	0.75
Pitch Preamplifier buffer amplifier	1	2.353	1.367	0.75
Yaw Preamplifier buffer amplifier	1	2.353	1.367	0.75
Attitude gyro power supply	1	1.13	6.305	0.75
Roll torque amplifier demodulator	1	2.659	4.002	0.75
Pitch torque amplifier demodulator	1	2.659	4.002	0.75
Yaw torque amplifier demodulator	1	2.659	4.002	0.75
Roll TC amplifier and temperature override	1	10.372	8.213	0.75
Pitch TC amplifier and temperature override	1	10.372	8.213	0.75
Yaw TC amplifier and temperature override	1	10.372	8.213	0.75

Table 4-15. Comparison of NAA/S&ID and Minneapolis-Honeywell Failure Rates (Continued)

Subassembly	Quantity	Failure Rates X 10^{-6} (hr)		Weight (lb) Each
		NAA	M-H	
Attitude error and switch	1	24.879	—	0.75
Gain network	1	0.108	6.174	0.75
Attitude error indicator	1	42.506	37.239	3.5
Attitude rate indicator	1	7.284	31.313	3.5
Attitude rate pitch	1	13.919	6.125	0.875
Attitude rate roll	1	13.919	6.125	0.875
Attitude rate yaw	1	13.919	6.125	0.875
Summing network	1	0.108	6.174	0.5
Total attitude indicator	1	420.771	79.869	3.5
Amplifier demodulator	3	2.754	1.614	0.6
+40 v d-c regulator (A)	1	0.731	—	0.6
Logic	3	4.146	9.972	0.6
Motor coil driver	3	1.728	2.559	0.6
Mode select logic	1	16.897	—	0.6
Resolver assembly	1	372.0	158.0	6.6
A-c and D-c rectifier	1	0.839	4.348	0.6
+20 v d-c regulator	1	1.33	0.607	0.6
Gimbal position indicator electronics	1	25.638	20.327	0.625

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Table 4-15. Comparison of NAA/S&ID and Minneapolis-Honeywell Failure Rates (Continued)

Subassembly	Quantity	Failure Rates X 10^{-6} (hr)		Weight (lb) Each
		NAA	M-H	
Delta velocity display	1	73.321	44.815	2.5
Delta velocity on-off electronics	1	32.35		0.88
Delta velocity ullage electronics	1	8.818	1.778	1.0
Attitude gyro roll demodulator	1	11.799	6.625	0.6
Attitude gyro pitch demodulator	1	11.799	6.625	0.6
Attitude gyro yaw demodulator	1	11.799	6.625	0.6
Roll d-c amplifier	1	6.982	2.249	0.6
Pitch d-c amplifier	1	6.982	2.249	0.6
Yaw d-c amplifier	1	6.982	2.249	0.6
Engine thrust electronics	1	0.976	2.498	0.6
Roll rate gyro demodulator	1	12.732	6.637	0.6
Pitch rate gyro demodulator	1	12.732	6.637	0.6
Yaw rate gyro demodulator	1	12.732	6.637	0.6
Roll mode select logic	1	16.897	10.045	0.6
Pitch mode select logic	1	16.897	14.756	0.6
Yaw mode select logic	1	16.897	16.668	0.6
Roll a-c and d-c power transformer	1	2.843	5.483	0.6
Pitch a-c and d-c power transformer	1	2.843	5.483	0.6

[redacted]

Table 4-15. Comparison of NAA/S&ID and Minneapolis-Honeywell Failure Rates (Continued)

Subassembly	Quantity	Failure Rates X 10^{-6} (hr)		Weight (lb) Each
		NAA	M-H	
Yaw a-c and d-c power transformer	1	2.843	5.483	0.6
Roll mode power relay	1	15.0	2.219	0.6
Pitch mode power relay	1	15.0	2.219	0.6
Yaw mode power relay	1	15.0	2.219	0.6
Roll d-c regulator	2	1.33	5.483	0.6
Pitch d-c regulator	2	1.33	5.483	0.6
Yaw d-c regulator	2	1.33	5.483	0.6
Yaw monitor and filter	1	4.946	7.048	0.6
Pitch monitor and filter	1	4.946	7.048	0.6
Yaw driver amplifier (A)	1	1.572	1.267	0.6
Yaw driver amplifier (B)	1	1.572	1.267	0.6
Pitch driver amplifier (C)	1	1.572	1.267	0.6
Pitch driver amplifier (D)	1	1.572	1.267	0.6
Yaw integrator	1	9.528	0.327	0.6
Pitch integrator	1	9.528	0.327	0.6
Yaw rate and servo demodulator	4	5.954	7.308	0.6
Pitch rate and servo demodulator	4	5.954	7.308	0.6
Attitude error electronics	1	14.0	16.289	0.875

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Table 4-15. Comparison of NAA/S&ID and Minneapolis-Honeywell Failure Rates (Continued)

Subassembly	Quantity	Failure Rates X 10^{-6} (hr)		Weight (lb) Each
		NAA	M-H	
Gimbal position indicator	1	31.906	22.237	2.0
Level detector	1	0.54	—	0.6
Minimum impulser	1	5.0	—	2.0
Rate gyro electronics	1	0.971	4.702	1.0
Roll rate gyro	1	150.0	100.0	0.62
Pitch rate gyro	1	150.0	100.0	0.62
Yaw rate gyro	1	150.0	100.0	0.62
Reaction jet driver	16	1.74	4.528	0.6
Back-up entry switch	1	48.8	0.09	1.95
Channel disable	1	48.8	1.2	1.95
Deadband adjust	1	24.2	0.03	1.95
Mode select	1	13.075	5.084	1.95
Translational control	2	4.15	1.05	9.0
Three-axis rotational control, direct	2	7.7	1.518	2.25
Three-axis rotational control, normal	2	27.8	7.862	9.6
Universal miscellaneous	1	24.176	5.072	0.942
Universal power supply	1	0.585	11.528	1.125

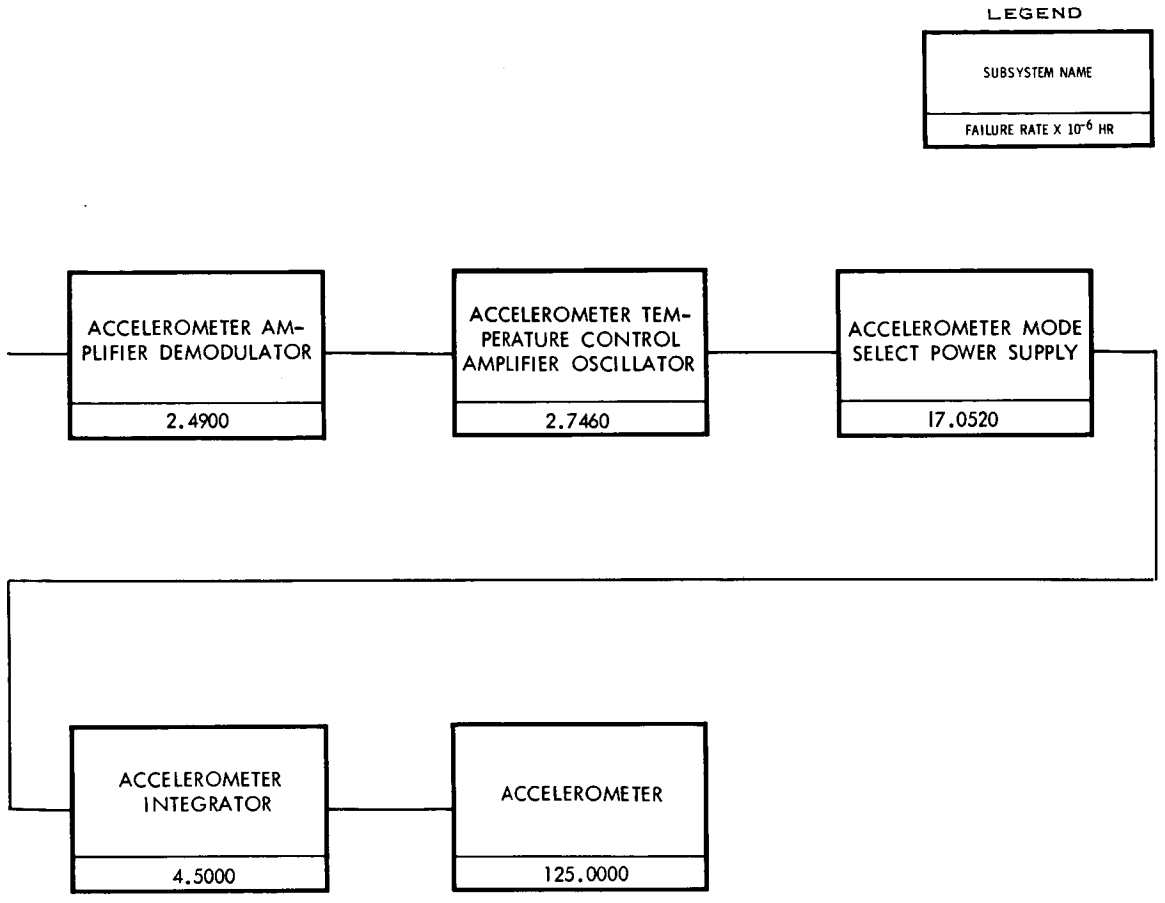

Figure 4-24. Accelerometer Package

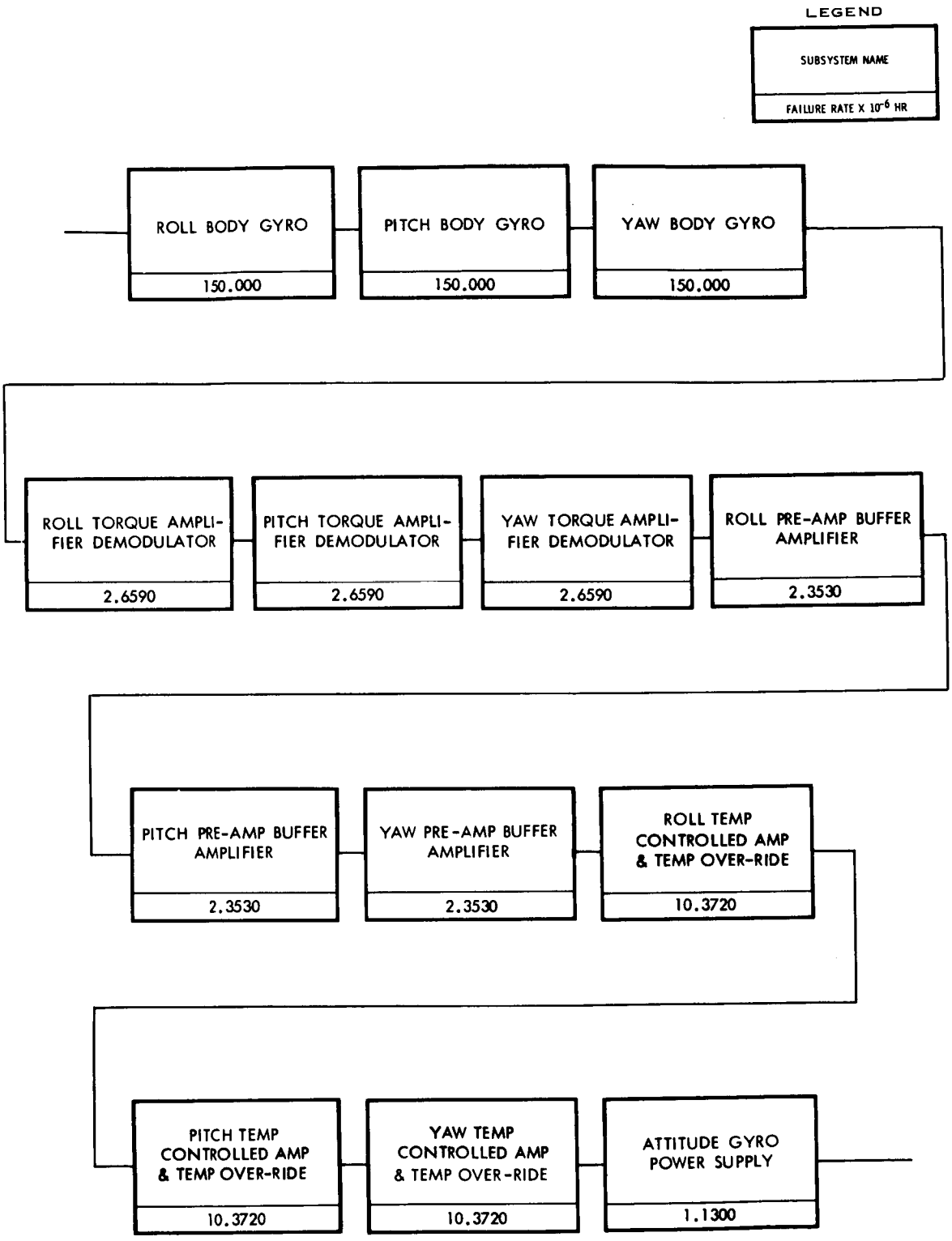

Figure 4-25. Body-Mounted Attitude Gyro

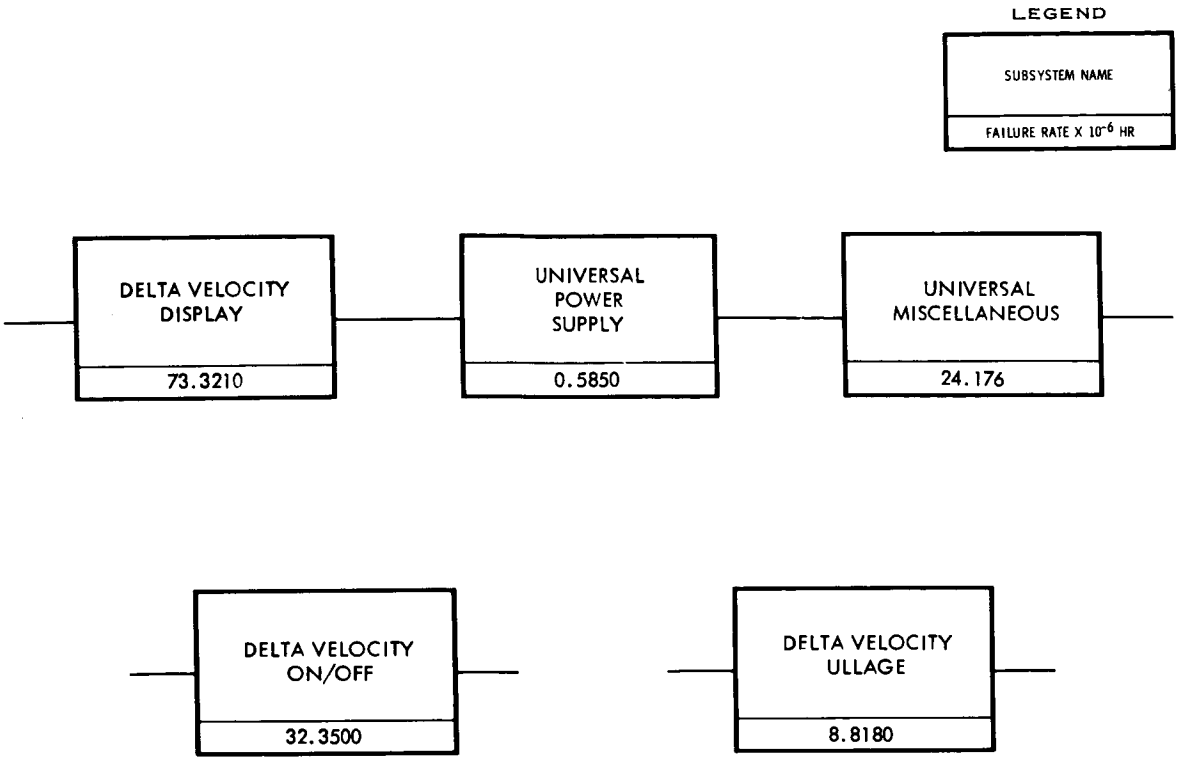

Figure 4-26. Delta Velocity Display

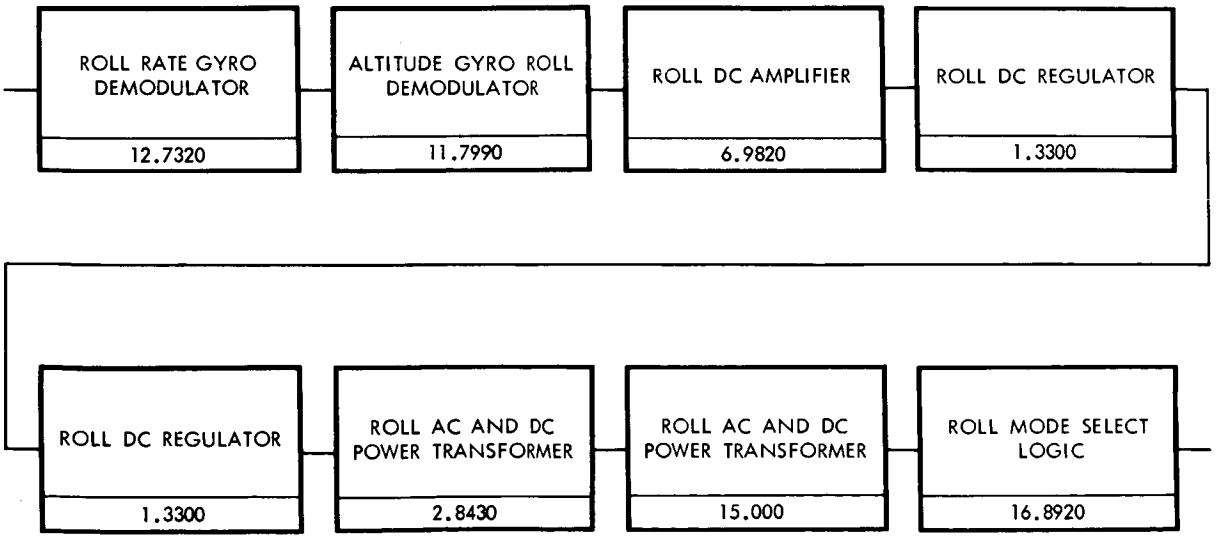

Figure 4-27. Electronic Control Assembly, Roll

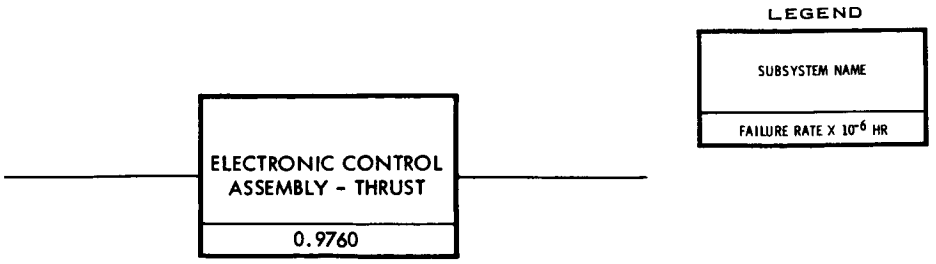

Figure 4-28. Electronic Control Assembly, Thrust

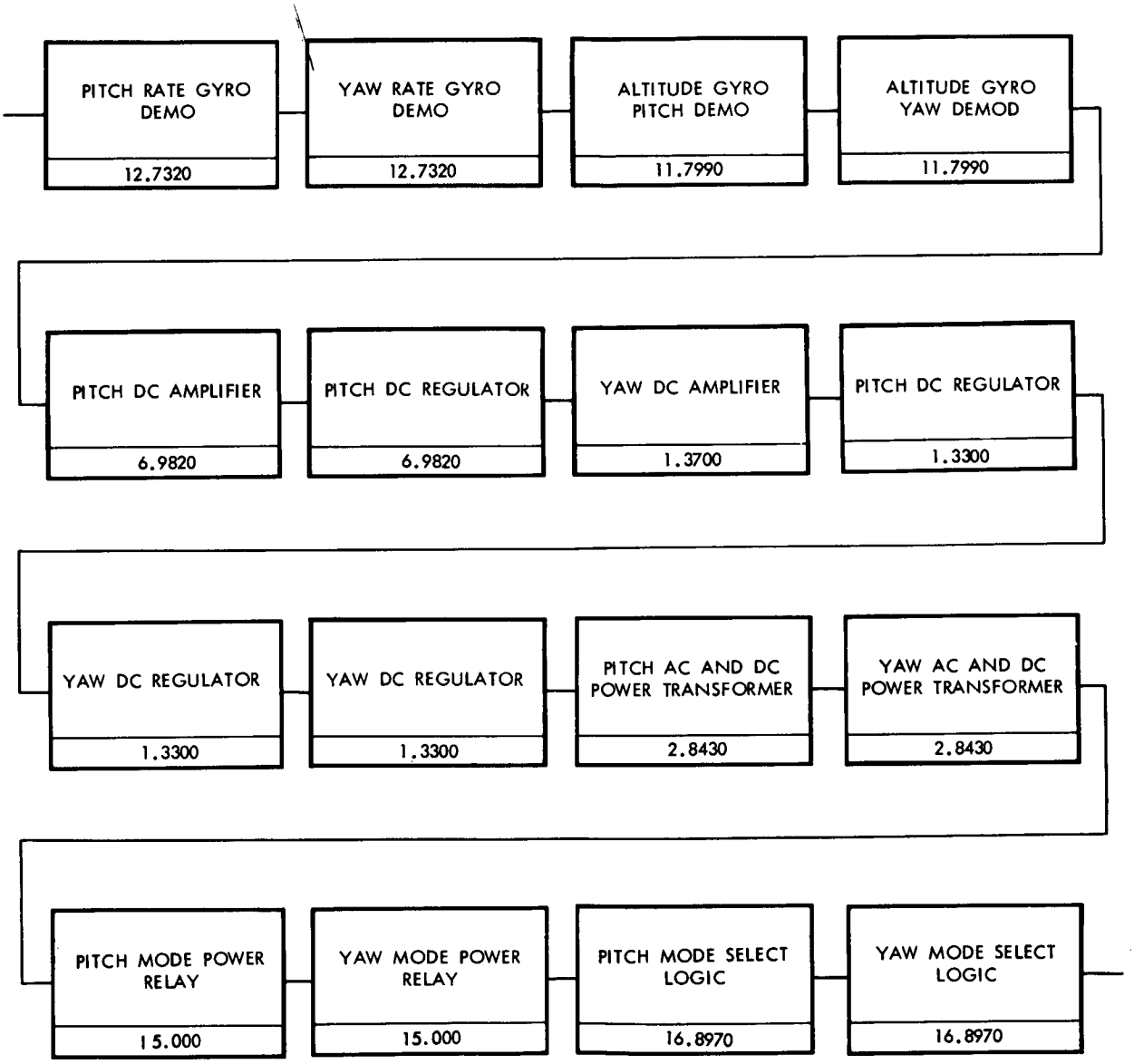

Figure 4-29. Electronic Control Assembly, Pitch Yaw

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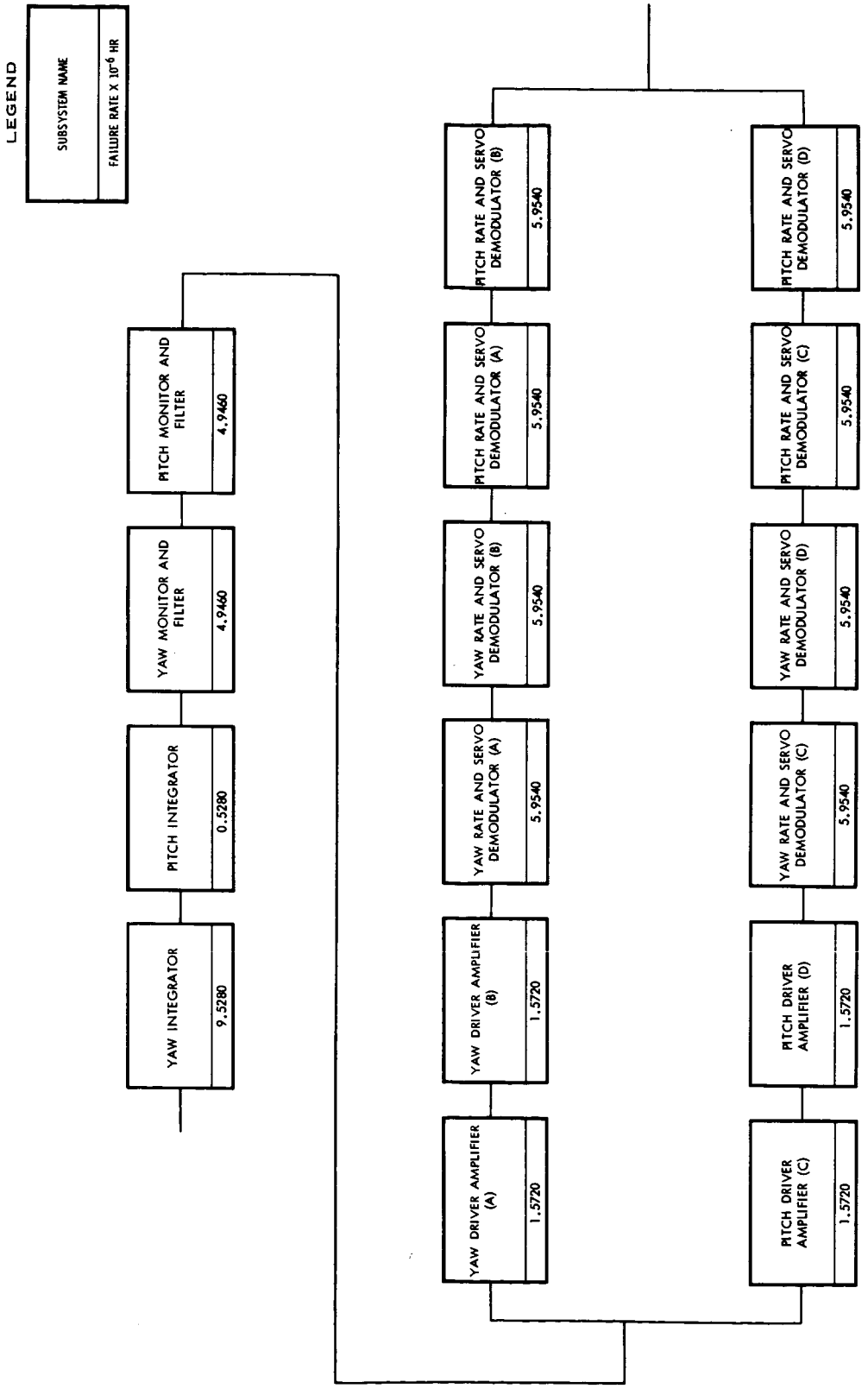

Figure 4-30. Electronic Control Assembly, Thrust Vector

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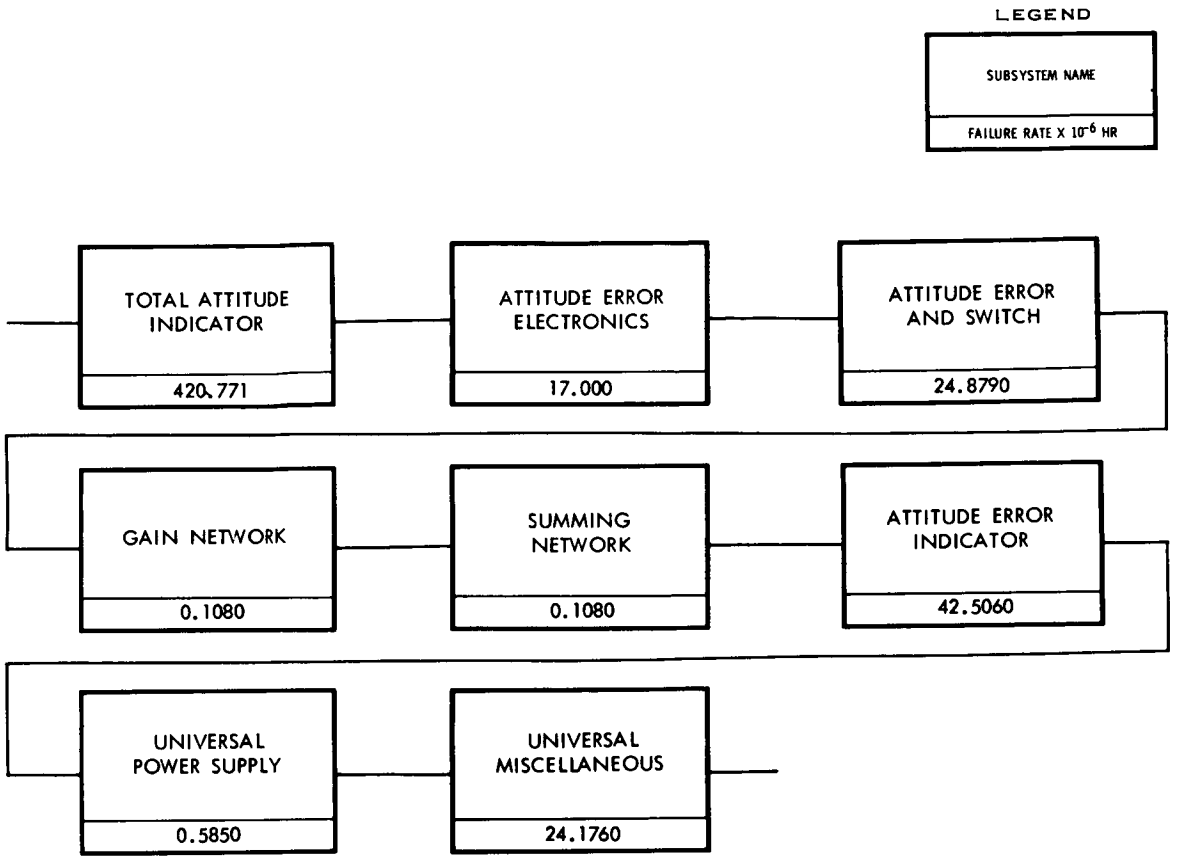

Figure 4-31. Flight Director Attitude Indicator

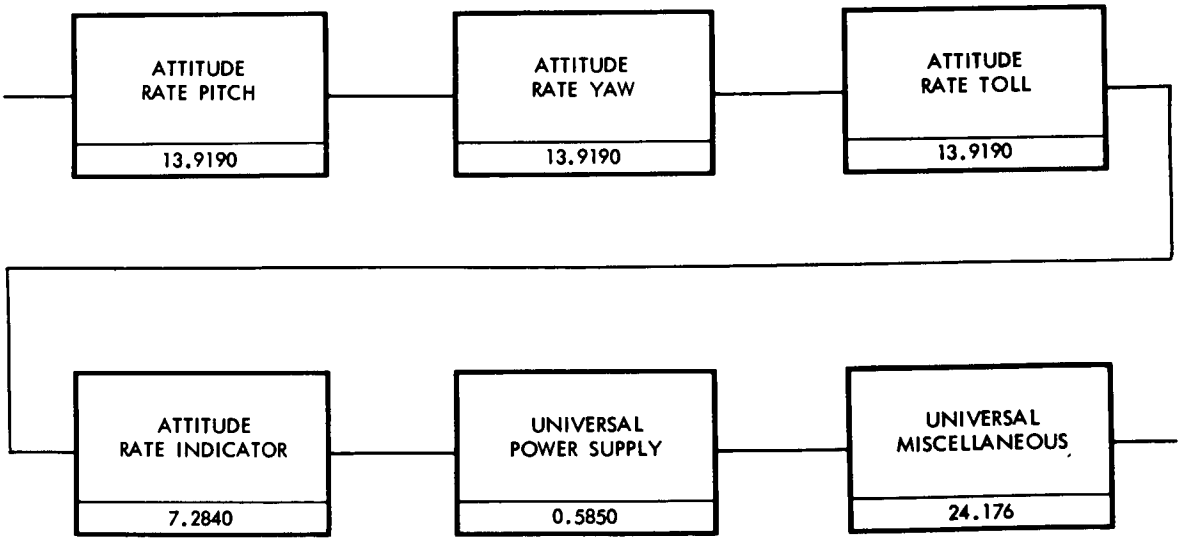

Figure 4-32. Flight Director Rate Indicator

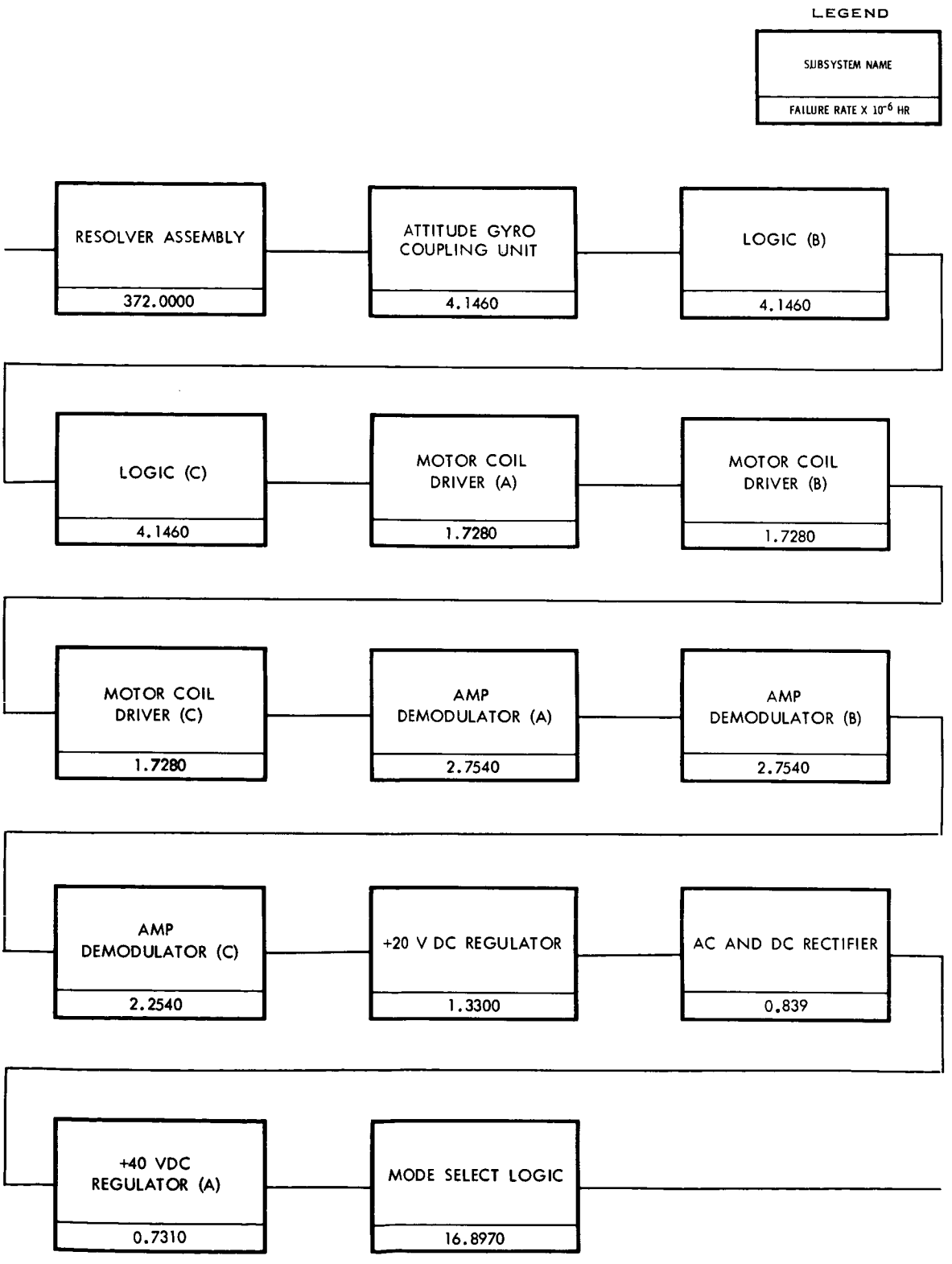

Figure 4-33. Attitude Gyro Coupling Unit

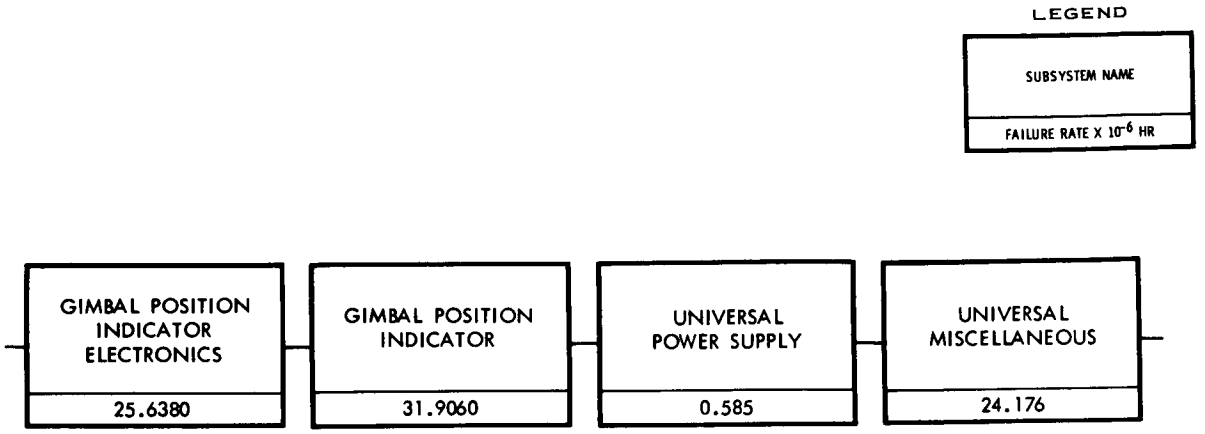

Figure 4-34. Gimbal Position Indicator

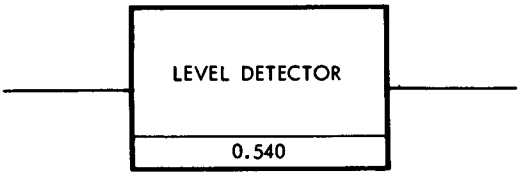

Figure 4-35. Level Detector

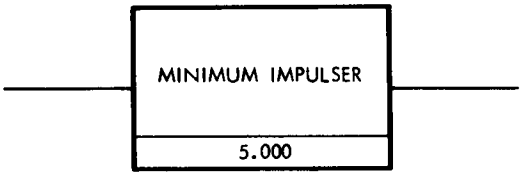

Figure 4-36. Minimum Impulser

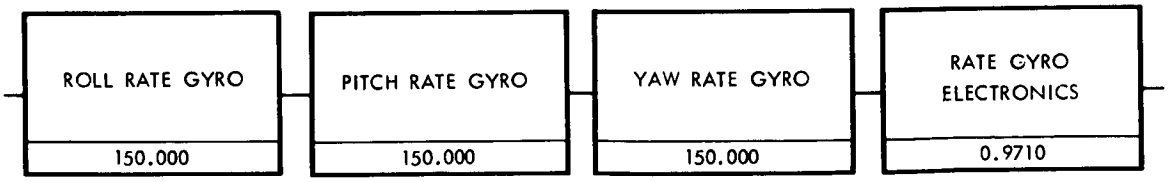

Figure 4-37. Rate Gyro Package

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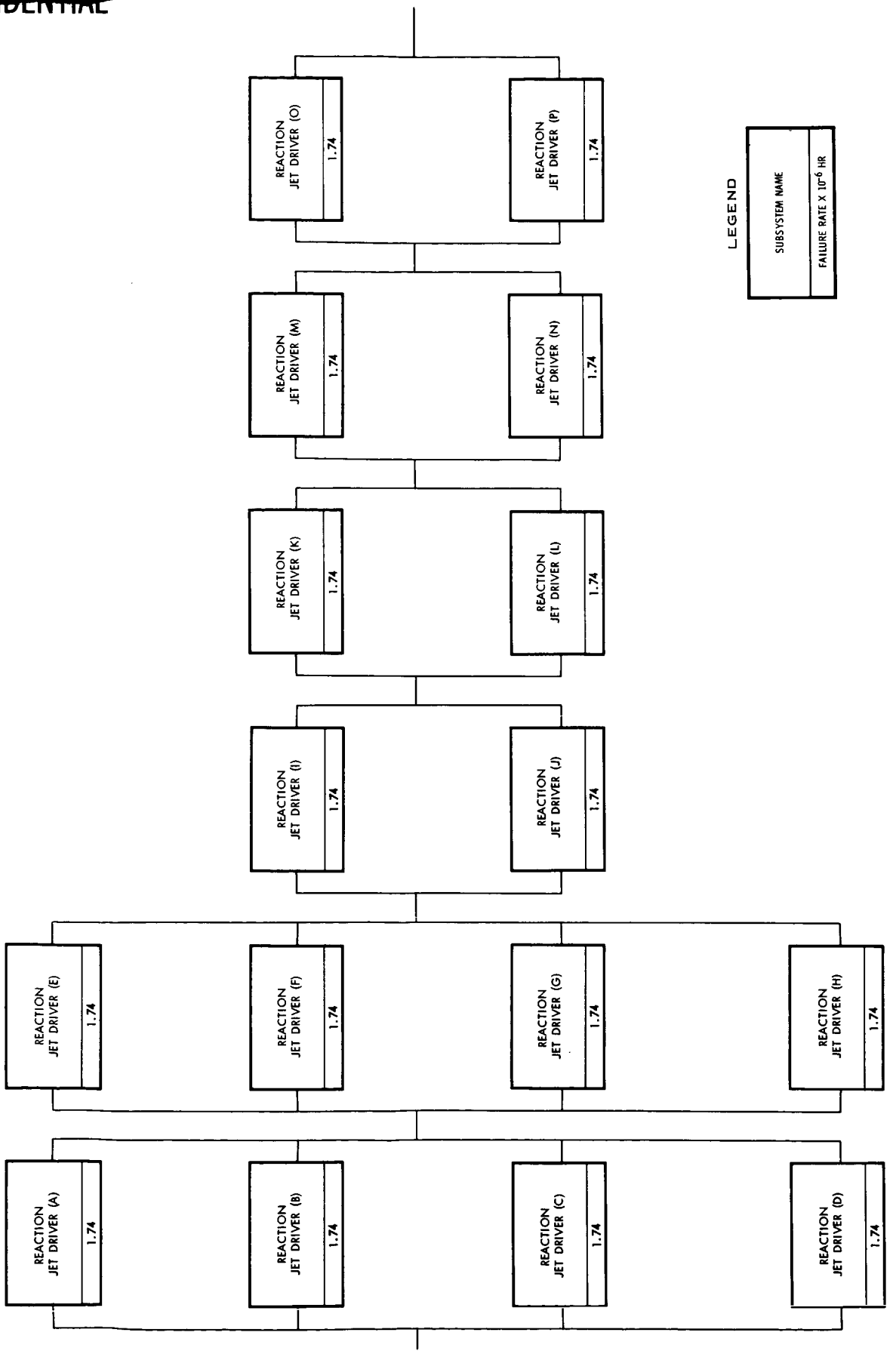
REACTION JET DRIVER (A) 1.74
REACTION JET DRIVER (B) 1.74
REACTION JET DRIVER (C) 1.74
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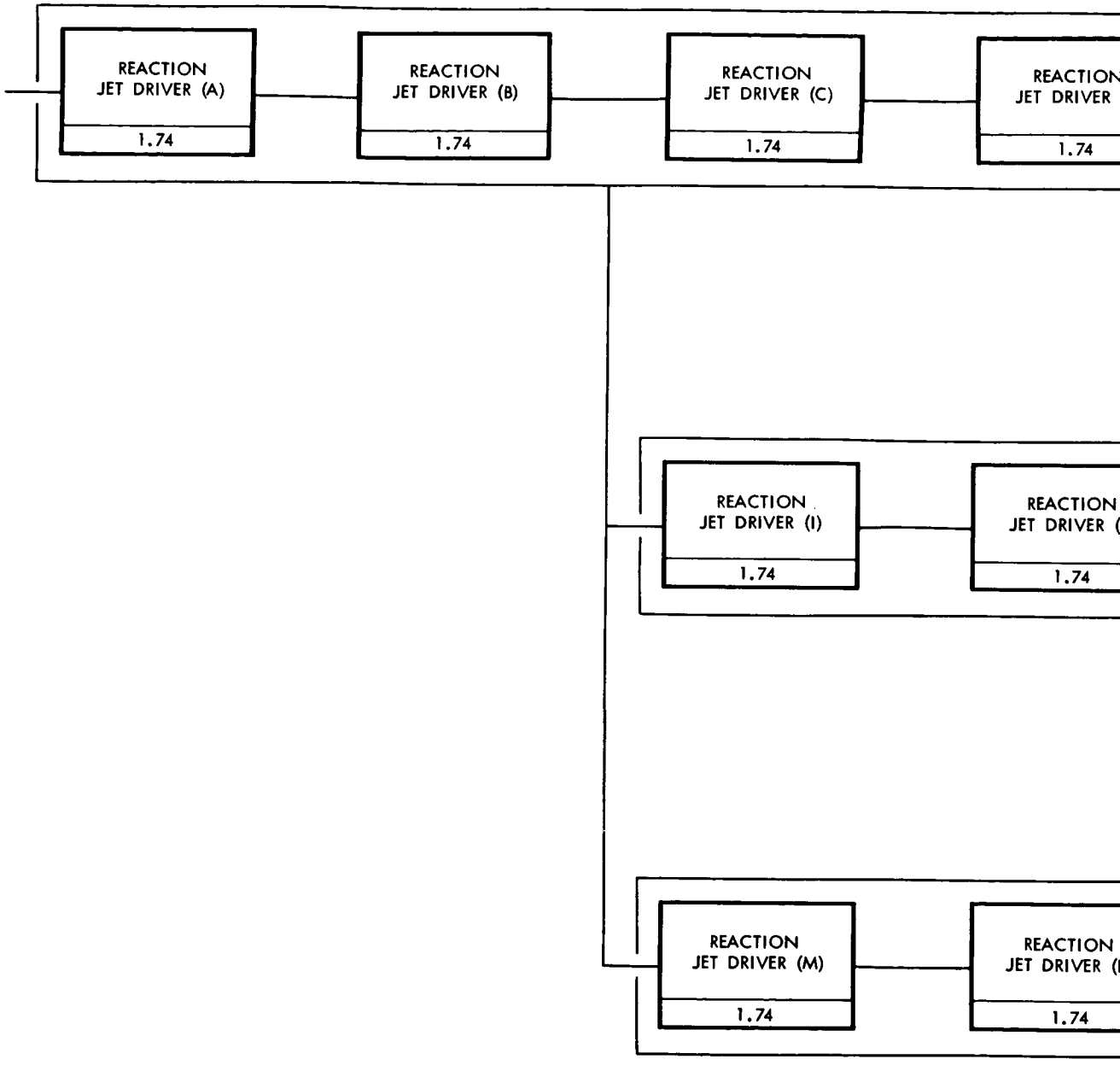
REACTION JET DRIVER (O) 1.74
REACTION JET DRIVER (P) 1.74
LEGEND
SUBSYSTEM NAME
FAILURE RATE X 10^{-6} HR

Figure 4-38. Reaction Jet Drivers, Command Module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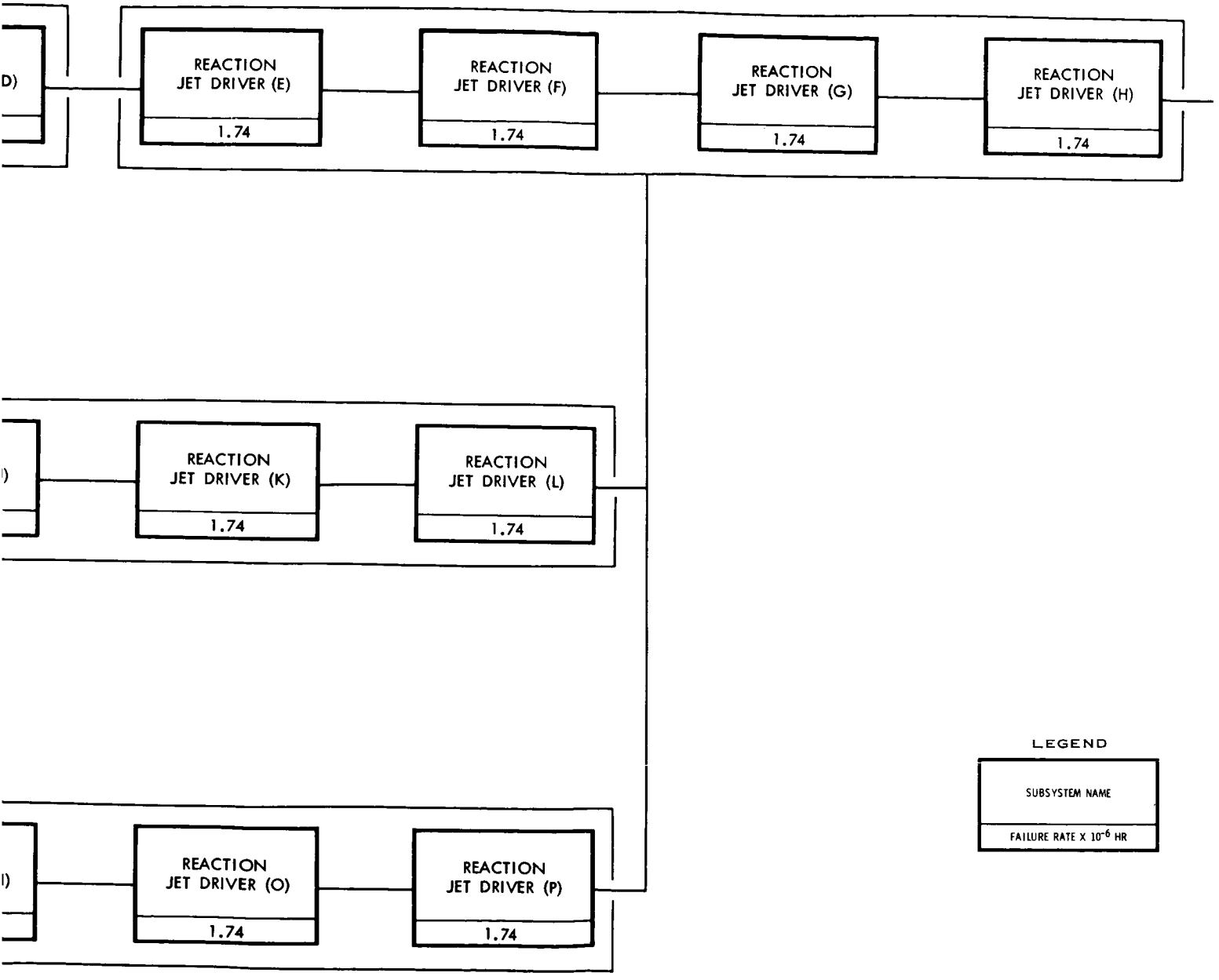

Figure 4-39. Reaction Jet Drivers, Service Module

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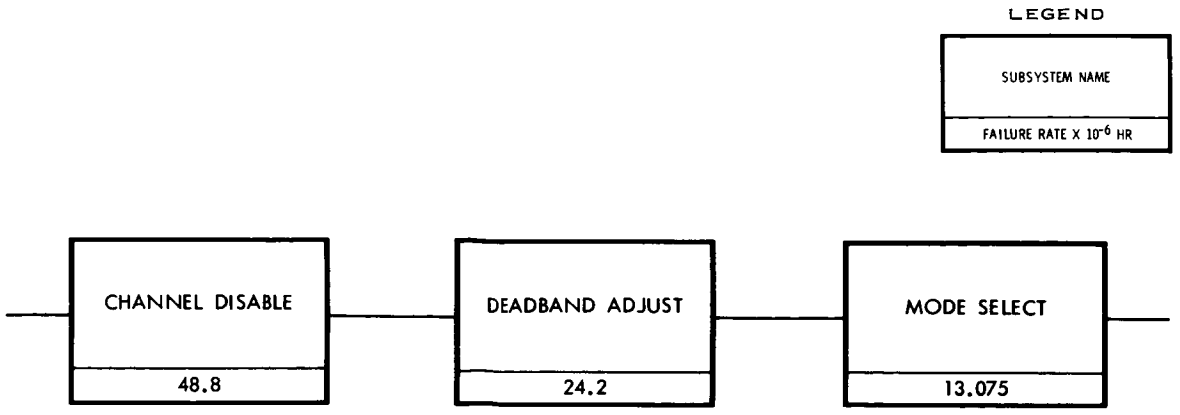


Figure 4-40. SCS Control Panel

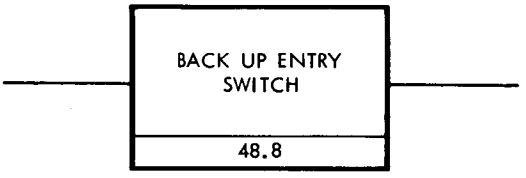


Figure 4-41. Back-Up Entry Switch

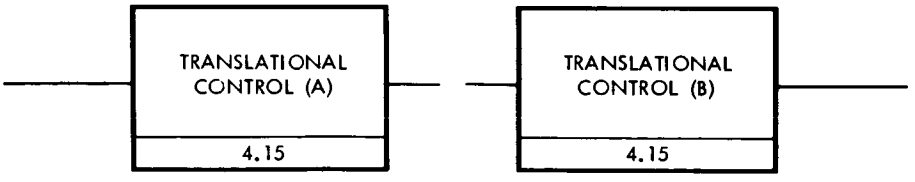


Figure 4-42. Translational Control

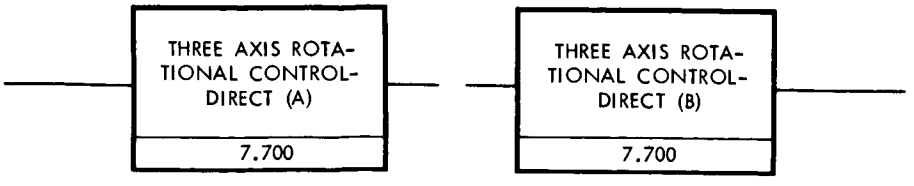


Figure 4-43. Three-Axis Rotational Control-Direct

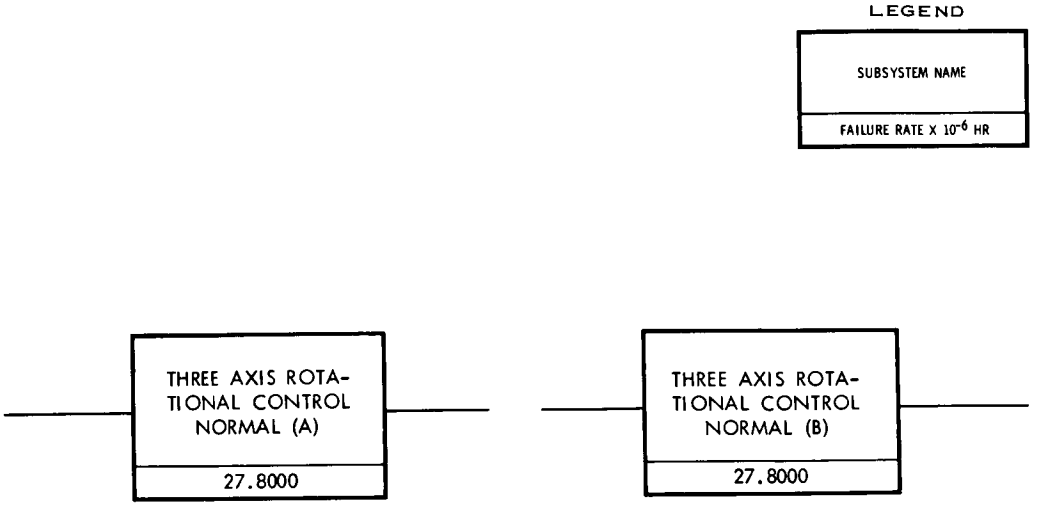


Figure 4-44. Three-Axis Rotational Control-Normal

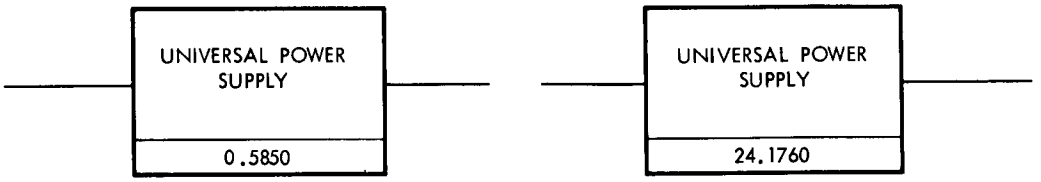


Figure 4-45. Universal Power Supply

Several thrust vector control servo-amplifier circuits of the ECA equipment have been analyzed to determine the redundancy required to achieve apportioned reliability requirements. The results indicate that part (transistor) redundancy does not allow the apportionment to be met. Table 4-16 shows the apportioned predicted (Apollo preferred), and part (transistor) redundnat reliabilities of the thrust vector control circuits, as analytically determined.

Table 4-16. Thrust Vector Control Circuit Apportionment and Predictions

Mission Success			
Amplifier	Apportioned Reliability	Predicted Reliability (Apollo preferred)	Reliability with Part (transistor) Redundancy
A4	0.9999998	0.9999964	0.9999989
A5, A6	0.9999998	0.9999966	0.9999992
A7, A8	0.9999999	0.9999969	0.9999995
Total	0.9999995	0.9999899	0.9999976

Table 4-17 shows the latest analytically determined reliability predictions of the SCS on the component level based upon updated parts count on the subassembly level. The operating times for the components were derived from logic diagrams and the Apollo LOR Mission Profile shown in Section II.

Failure Effects Analysis

The results of preliminary failure effects analysis of the stabilization and control subsystem at the equipment level are given in Table 4-20 at the end of this section.

Table 4-17. Stabilization and Control Subsystem Component Level Reliability Predictions

Component	Time (t) Hours	Number (n) of Components X Failure Rate X 10^{-6} (hr)	Time X(n)X Failure Rate	Mission Success Reliability
Electronic Control Assembly Pitch, Yaw axes	336.0	88.78	29830.08	0.970639
Electronic control assembly, Roll axis	336.0	44.39	14915.04	0.985210
Reaction jet drivers (SM)	335.0			0.999999
Reaction jet driver (CM)	0.333			0.999999
Thrust vector control	5.6			0.999999
Translational control	61.8			0.999999
Three-axis rotational control, normal	61.8			0.999999
Three-axis rotational control, direct	40.0			0.999999
Engine thrust	0.14	0.61	0.085	0.999999
Body-mounted attitude gyros	336.0	187.46	62986.56	0.939037
Attitude gyro coupling unit	336.0	268.26	90135.36	0.913839
Rate gyro package	316.0	153.13	51451.68	0.949804
Accelerometer package	336.0	67.66	22733.76	0.977556
Minimum impulser	125.12	5.0	625.60	0.999374
Level detector	0.333	0.54	0.1798	0.999999

Table 4-17. Stabilization and Control Subsystem Component Level Reliability Predictions (Continued)

Component	Time (t) Hours	Number (n) of Components X Failure Rate X 10^{-6} (hr)	Time X(n)X Failure Rate	Mission Success Reliability
Flight director attitude indicator	310.0	1143.08	354354.80	0.701594
Flight director rate indicator	2.0	24.06	48.12	0.999952
Delta velocity display	20.0	47.20	944.0	0.999056
Delta velocity on-off	0.14	21.17	2.964	0.999997
Delta velocity ullage	3.0	5.66	16.98	0.999983
Gimbal position indicator	5.6	20.61	115.416	0.999884
Universal display electronics	310.0	18.06	5598.6	0.994401
SCS control panel	336.0	57.46	19306.56	0.980885
Back-up entry switch	0.333	32.56	10.842	0.999989
Total system				0.520486

SUBSYSTEM DESCRIPTION

A typical reliability logic diagram of the SCS broken down to the subassembly level for the navigational sighting function, is shown in Figure 4-46.

The diagram shows the series constraints of the subsystem and the redundancy on the subassembly level. The operational modes are the basis of this configuration. They are as follows:

1. A signal is generated from the guidance and navigation subsystem and sent into the SCS through the ECA pitch, roll and yaw, which is comprised of the following series subassemblies: roll, pitch, yaw, demodulators; roll, pitch, yaw dc amplifiers; and roll, pitch, yaw

mode select logic. For these subassemblies to function, the following series elements must also function for power supply and regulation: roll, pitch, yaw ac and dc power transformers; roll, pitch and yaw mode power relays; and roll, pitch and yaw dc regulators. These subassemblies perform an attitude hold function in conjunction with the input from the rate gyro package consisting of the following series elements: rate gyros roll, pitch, yaw; and rate gyro electronics. Rate gyro inputs can be backed up by the body mounted attitude gyros when they are in the rate mode and consist of the following series elements: roll, pitch and yaw gyros; roll, pitch and yaw pre-amplifier buffers; attitude gyro power supply; roll, pitch, yaw torque amplifier demodulators; and the roll, pitch and yaw temperature control amplifier and temperature override. After the signal goes through limiting and other circuitry, it culminates in the reaction jet drivers which in turn drive the reaction jets of the RCS. The spacecraft attitude can be kept in a 5 or 0.5 degree deadband in any of the three axis.

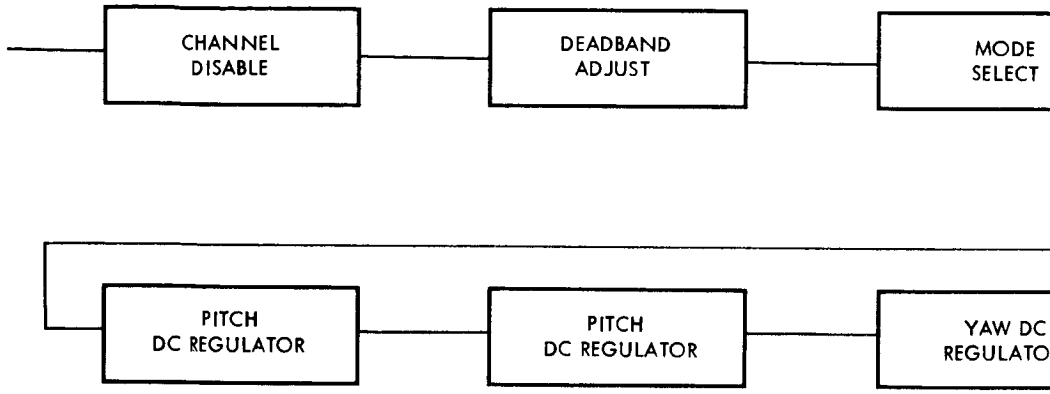
2. This function can also be accomplished by switching to the SCS hold mode which requires the use of the SCS control panel. The panel is comprised of the following series elements: channel disable, deadband adjust, and mode select. This places the body mounted attitude gyros in the attitude mode and replaces the signal generated from the navigation and guidance subsystem. In conjunction with the body mounted attitude gyros, the following series elements are also needed: roll, pitch, yaw amplifier demodulators; roll, pitch, yaw logic; +40 volts dc regulator; +21 volts dc regulator; resolver assembly; motor-coil drivers; mode select logic; and ac and dc rectifier. Again in this mode all the electronic control subassemblies must be included in the series configuration.

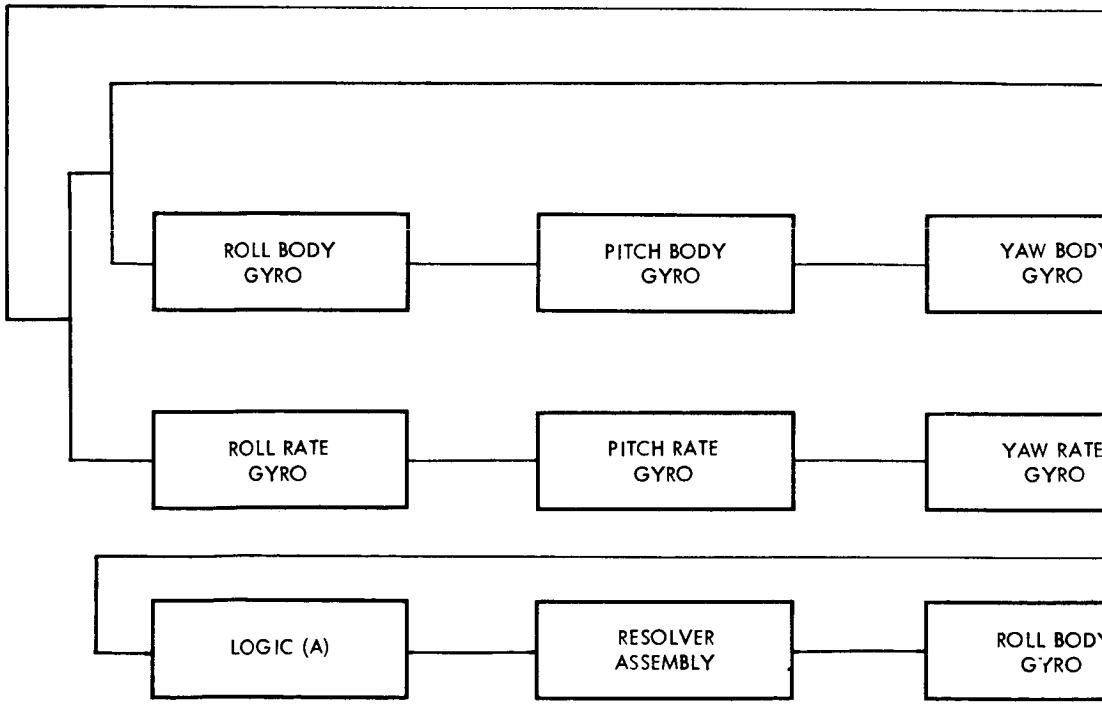
3. In order to rotate the spacecraft, either of the two three-axis rotational controls are utilized.

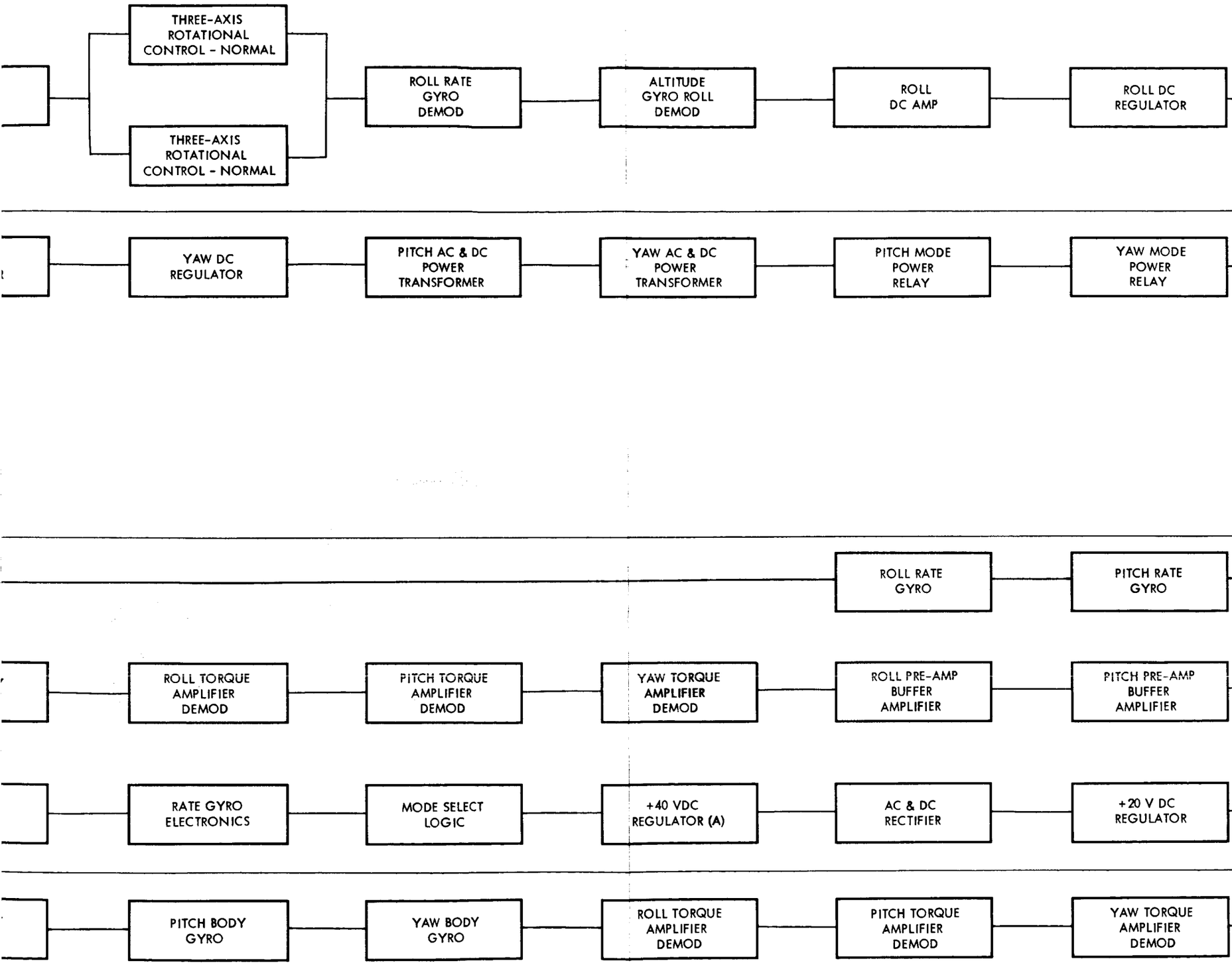
A brief description of all SCS equipment is shown in Table 4-18.

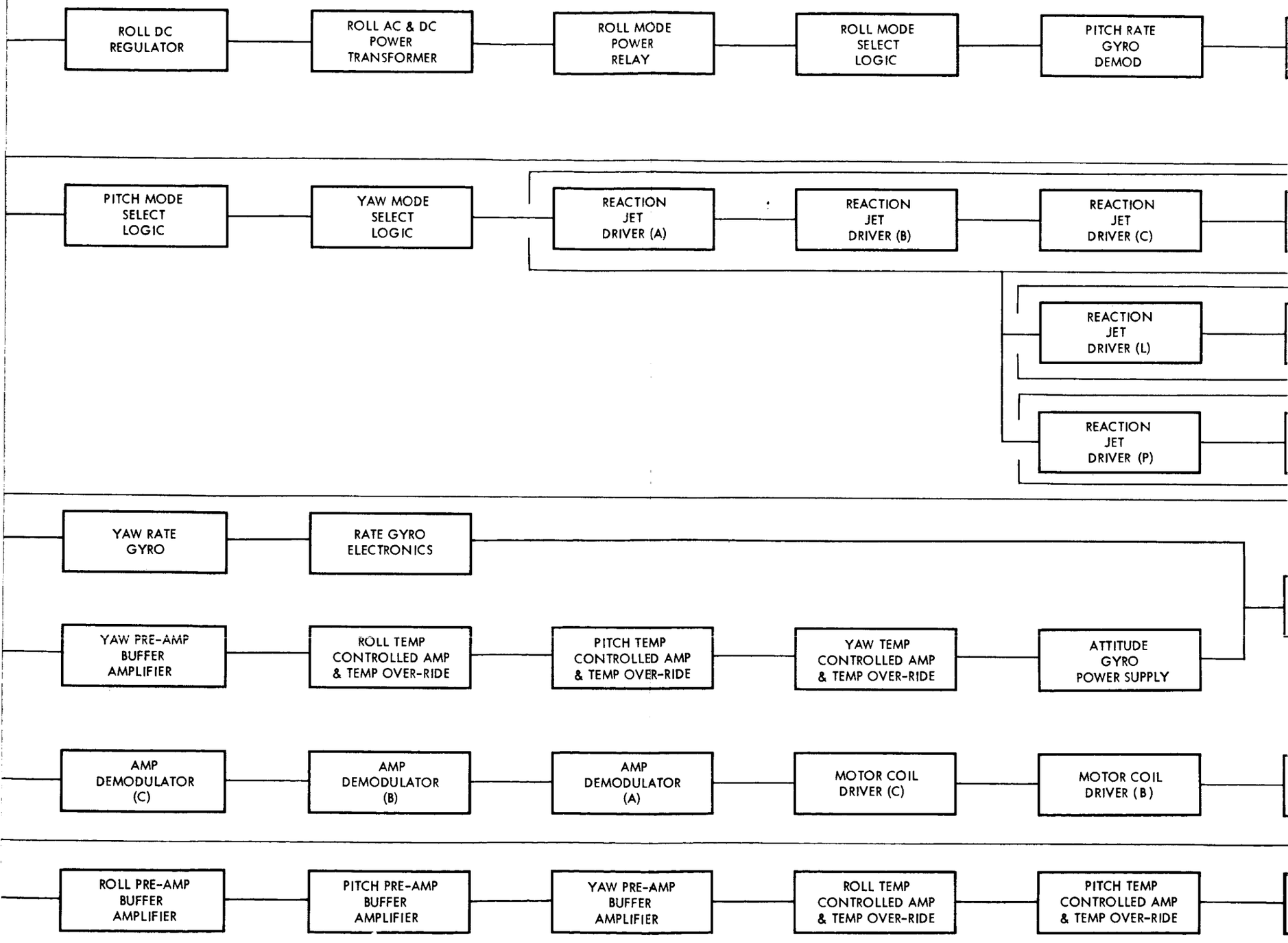
SPECIAL STUDIES

NAA/S&ID and Minneapolis-Honeywell have completed preliminary in-flight spare studies. These studies were conducted with the realization of the spare weight restraints and the mission success reliability requirements. Certain specific areas of the over-all reliability have shown definite improvement as compared to previous estimates. This is due to the inherent redundancy of functioning subassemblies within the stabilization and control subsystem.


CHANNEL DISABLE
DEADBAND ADJUST
MODE SELECT
PITCH DC REGULATOR
PITCH DC REGULATOR
YAW DC REGULATO


ROLL BODY GYRO
PITCH BODY GYRO
YAW BODY GYRO
ROLL RATE GYRO
PITCH RATE GYRO
YAW RATE GYRO
LOGIC (A)
RESOLVER ASSEMBLY
ROLL BODY GYRO


THREE-AXIS ROTATIONAL CONTROL - NORMAL
THREE-AXIS ROTATIONAL CONTROL - NORMAL
ROLL RATE GYRO DEMOD
ALTITUDE GYRO ROLL DEMOD
ROLL DC AMP
ROLL DC REGULATOR
YAW DC REGULATOR
PITCH AC & DC POWER TRANSFORMER
YAW AC & DC POWER TRANSFORMER
PITCH MODE POWER RELAY
YAW MODE POWER RELAY
ROLL RATE GYRO
PITCH RATE GYRO
ROLL TORQUE AMPLIFIER DEMOD
PITCH TORQUE AMPLIFIER DEMOD
YAW TORQUE AMPLIFIER DEMOD
ROLL PRE-AMP BUFFER AMPLIFIER
PITCH PRE-AMP BUFFER AMPLIFIER
RATE GYRO ELECTRONICS
MODE SELECT LOGIC
+40 VDC REGULATOR (A)
AC & DC RECTIFIER
+20 V DC REGULATOR
PITCH BODY GYRO
YAW BODY GYRO
ROLL TORQUE AMPLIFIER DEMOD
PITCH TORQUE AMPLIFIER DEMOD
YAW TORQUE AMPLIFIER DEMOD


ROLL DC REGULATOR
ROLL AC & DC POWER TRANSFORMER
ROLL MODE POWER RELAY
ROLL MODE SELECT LOGIC
PITCH RATE GYRO DEMOD
PITCH MODE SELECT LOGIC
YAW MODE SELECT LOGIC
REACTION JET DRIVER (A)
REACTION JET DRIVER (B)
REACTION JET DRIVER (C)
REACTION JET DRIVER (L)
REACTION JET DRIVER (P)
YAW RATE GYRO
RATE GYRO ELECTRONICS
YAW PRE-AMP BUFFER AMPLIFIER
ROLL TEMP CONTROLLED AMP & TEMP OVER-RIDE
PITCH TEMP CONTROLLED AMP & TEMP OVER-RIDE
YAW TEMP CONTROLLED AMP & TEMP OVER-RIDE
ATTITUDE GYRO POWER SUPPLY
AMP DEMODULATOR (C)
AMP DEMODULATOR (B)
AMP DEMODULATOR (A)
MOTOR COIL DRIVER (C)
MOTOR COIL DRIVER (B)
ROLL PRE-AMP BUFFER AMPLIFIER
PITCH PRE-AMP BUFFER AMPLIFIER
YAW PRE-AMP BUFFER AMPLIFIER
ROLL TEMP CONTROLLED AMP & TEMP OVER-RIDE
PITCH TEMP CONTROLLED AMP & TEMP OVER-RIDE

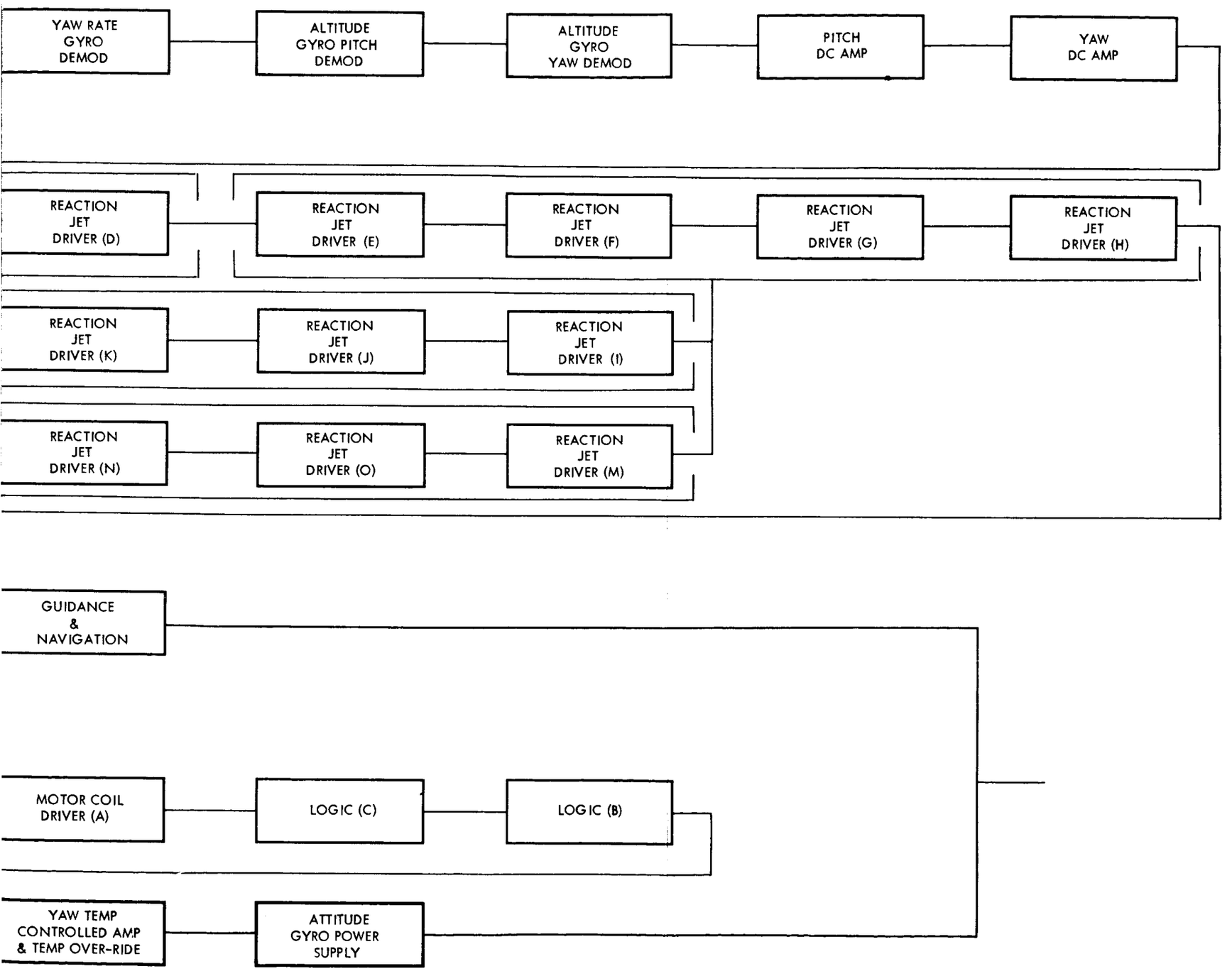


Figure 4-46. Stabilization and Control Subsystem Navigational Sighting Logic Diagram

Table 4-18. Stabilization and Control Subsystem Equipment

Equipment	Function
Rate gyro package	Contains sensing elements and circuitry to provide and display angular rate stabilization.
Attitude gyro, accelerometer package	Provides angular displacement signals for control and display. The body mounted accelerometer will provide x-axis acceleration data for delta velocity display and entry switching.
SCS control panel	Provides selection of any of the spacecraft control modes to adjust the deadband, to disable and re-enable the SCS outputs by channels to select the attitude gyros as a back-up rate source by channel, and to select the post 0.05-g switching configuration during entry modes.
Electronic control assembly, pitch axis	Contains circuitry for resolving, summing, shaping, and switching of the pitch channel sensors and manual input signals required to control the operation of the pitch reaction jets for the service module and the command module. The pitch channel circuitry also controls the pitch channel of the propulsion engine thrust vector circuits for the service module, and will provide pitch channel circuit isolation and signal conditioning elements required for ground and in-flight checkout, monitoring, and telemetry.
Electronic control assembly, roll axis	Contains circuitry for resolving, shaping, and switching of the roll channel sensors and manual input signals required to control the operation of the roll reaction jets for the service and the command modules. The roll channel will also provide roll channel circuit isolation and signal conditioning elements required for ground and in-flight checkout, monitoring, and telemetry.

Table 4-18. Stabilization and Control Subsystem Equipment (Continued)

Equipment	Function
Electronic control assembly, yaw axis	Contains circuitry for resolving, summing, shaping, and switching of the yaw channel sensors and manual input signals required to control the operation of the yaw reaction jets for the service and command modules. The yaw channel circuitry also controls the yaw channel of the propulsion engine thrust vector circuits for the service module, and provides yaw channel circuit isoluation and signal conditioning elements required for ground and in-flight checkout, monitoring, and telemetry.
Electronic control assembly, auxiliary	Contains circuitry and components to provide propulsion engine on-off control, mode select logic, signal conditioning, attitude gyro coupling, and spin motor rotation detection.
Electronic control assembly, display and attitude gyro and accelerometer electronics	Contains circuitry and components for the flight director attitude indicator, delta velocity display, and the attitude set and gimbal position display. Circuitry for the body-mounted attitude gyros and the x-axis accelerometer will also be included.
Flight director attitude indicator	Displays spacecraft altitude and attitude errors from the Guidance and Navigation and SCS subsystems and angular rate in formation for manual control and monitoring purposes.
Attitude set and gimbal position display	Provides circuitry and components to enable the pilot to align the attitude indicator during SCS local vertical or SCS attitude control modes, to set and insert Euler angles so that the differences between these angles and the attitude gyro coupling unit shaft positions appear on the attitude error indicators as body-referenced error angles, to display the

Table 4-18. Stabilization and Control Subsystem Equipment (Continued)

Equipment	Function
Attitude set and gimbal position display (Continued)	angular position of the pitch and yaw SPS engine gimbals, and to provide a manual control device for engine gimbal positioning.
Delta velocity display	Displays delta velocity magnitude remaining for pilot monitoring and control purposes, manually inserts the desired delta velocity magnitude into the display electronics package, displays and initiates thrust on and off signals to the SPS engine, inserts tail-off corrections, and initiates direct ullage commands.
Translational control	Provides individual or simultaneous outputs for ±X, ±Y, and ±Z acceleration commands without cross coupling and provides a signal for manual SPS thrust-off command.
Three-axis rotational control	Provides simultaneous multi-axis actuation without cross coupling, provides proportional rate commands for ±p, ±q and ±r with a threshold command of 0.1 degree per second and a maximum command of 17 degrees per second, provides direct commands for $\pm\dot{p}$, $\pm\dot{q}$ and $\pm\dot{r}$, and provides switching signals by channel to indicate with i the SCS electronics presence of either normal or emergency commands.

Table 4-19 shows a preliminary list of the subassemblies spared and their weight. This is a result of the first preliminary spares study. This study will continue until a reliability estimate has been made showing the results of the spares study.

Table 4-19. Preliminary Subassembly Spares List

Component	Subassembly	Number	Weight (lb)
Delta velocity display	Delta velocity on-off electronics	2	1.75
	Delta velocity ullage electronics	2	2.00
Electronic control assembly, roll; electronic control assembly, pitch yaw	Attitude gyro demodulator	1	0.60
	D-c amplifier	1	0.60
	Rate gyro demodulator	1	0.60
	Mode select logic	1	0.60
	A-c and d-c power transformer	1	0.60
	Mode power relay	1	0.60
	D-c regulator	1	0.60
Electronic control assembly, thrust vector	Monitor and filter	4	2.40
	Integrator	4	2.40
Attitude gyro coupling unit	Amplifier demodulator	1	0.60
	+40 v d-c regulator	1	0.66
	Logic	1	0.60
	Motor coil driver	1	0.60
	Mode select logic	1	0.60
	Resolver assembly	1	0.60
	A-c and d-c rectifier	1	0.60
	+20 dc regulator	1	0.60

Table 4-19. Preliminary Subassembly Spares List (Continued)

Component	Subassembly	Number	Weight (lb)
Electronic control assembly, thrust	Engine thrust electronics	4	2.40
Gimbal position indicator	Gimbal position indicator electronics	2	1.25
	Gimbal position indicator	2	4.00
Minimum impulser	Minimum impulser	1	2.00
Rate gyro package	Rate gyro	2	1.24
	Rate gyro electronics	1	1.00
Accelerometer package	Accelerometer	1	0.62
	Accelerometer amplifier demodulator	1	0.75
	Accelerometer integrator	1	1.50
	Accelerometer mode select power supply	1	1.50
	Accelerometer temperature control amplifier oscillator	1	0.75
Body-mounted attitude gyro	Body gyro	2	2.40
	Preamplifier buffer amplifier	2	1.50
	Attitude gyro power supply	2	1.50
	Torque amplifier demodulators	2	1.50
	Temperature-controlled amplifier and temperature override	2	1.50
Delta velocity display	Delta velocity display	2	5.00

Table 4-19. Preliminary Subassembly Spares List (Continued)

Component	Subassembly	Number	Weight (lb)
SCS control panel	Channel disable	2	3.90
	Deadband adjust	2	3.90
	Mode select	2	3.90
Universal power supply	Universal miscellaneous	3	2.83
	Universal power supply	3	3.38

PROBLEM AREAS

Sparing Versus Redundancy

Because of the criticality of the thrust vector-control circuitry, a configuration analysis is being conducted. Several studies of the component's reliability have already been made. Circuit design has not yet been finalized. Some aspects under scrutiny are operational redundancy versus sparing and which subassembly requires redundancy or sparing. The impossibility of putting the crew in the loop during a critical boost phase for sparing operations is a severe constraint in this study.

Gyros

Gyros in general are relatively unreliable. A study of how to bring the gyro components up to the reliability of the series electrical element is now progressing. Such concepts as that of gyro failure modes and duty-cycle effect on reliability are being studied in order to achieve a realistic solution.

Relays

Because of the size of relays and their high failure rate, a problem area has arisen as a result of their proposed use in the SCS control panel. Standard size relays are not compatible with the volume allocation for the panel, and therefore, smaller relays are required. Studies are now in progress to incorporate this type of relay. Because of the required reliability of the item, a method for testing the validity of reliability data must be found.

TEST PROGRAM

Qualification Test Program

The qualification program was finalized during the quarter. Three sets of SCS hardware will be tested at the component or black box level for the design proof tests. Two sets will be refurbished for off-limit testing. Three systems will be tested for wearout for the life test phase.

Test Program Status

Program Development

A detailed evaluation of the Minneapolis-Honeywell cost proposal was conducted during the quarter, and a base for contract negotiations was established. Particular attention was given to possible duplication of effort. Insurance of development and qualification test efforts were funded as design and development costs rather than reliability costs. Contract negotiations with Minneapolis-Honeywell will be completed during the following quarter.

Specific Development Tests

Breadboard development tests of operational and switching amplifiers, reaction jet drivers, thrust vector control amplifiers, power supply, flip-flop circuits and DV oscillators to determine stability and performance under both ambient and temperature conditions were conducted during the quarter. The testing necessitated design modifications in some areas to achieve desired performance.

Life tests on components of the miniature integrating attitude gyros were initiated during the quarter. The spin motor running detector and the dualsyn have already completed six weeks of testing without any failures being reported. These development tests will determine the gyro's life expectancy.

Life tests on FDAI components were also initiated. A slip ring assembly completed 150,000 cycles without failure and 5 servo amplifiers have successfully completed ambient and temperature tests.

The following number of materials have already completed the materials test program: gassing tests, 154; flammability tests, 78; high

temperature aging, 108, and vacuum testing, 34. This testing will be continued during the next quarter. Parameter variation tests of critical circuits and application suitability tests of electronic parts will also be continued.

SUBCONTRACTOR MANAGEMENT

The spares review team met in December, 1962 to determine the required SCS spares to support the BP missions. Because of the criticality of spares support for BP's 14 and 22 and AFRM 001, the team was asked to consider sparing the Minneapolis-Honeywell SCS system. Due to the lack of part numbers, parts under consideration will have to be shop-released for procurement by nomenclature only.

A meeting of the SCS cost proposal review team concerned with analysis and fact finding was held in January, 1963. Discussions were held on Minneapolis-Honeywell general program ground rules, which were set for the purpose of implementing an funding cost proposal. Such discussions were limited to Minneapolis-Honeywell document A62 776B4(1), Program Basis.

A detailed evaluation of the Minneapolis-Honeywell cost proposal dealing with the use of high reliability parts in the SCS subsystem has been completed. A listing of part types and estimated costs between MIL-Standard and Minuteman (or equivalent) parts for the SCS equipments was established. From this evaluation NAA/S&ID will negotiate a firm cost proposal based upon the use of high reliability parts.

NASA/MSC and NAA/S&ID representatives met in February, 1963 to negotiate costs of the SCS system. NAA/S&ID and Minneapolis-Honeywell cost estimates were also discussed. NASA/MSC requested that a different analysis scheme on the estimate be performed prior to firm contract negotiations with Minneapolis-Honeywell.

NAA/S&ID Reliability representatives also met with Minneapolis-Honeywell Reliability personnel in February to discuss the evaluation of SCS requirements in terms of mission success and safe abort objectives. The clarification of mission success and safe abort was also discussed.

PLANNED ACTIVITIES

NAA/S&ID has directed the SCS system subcontractor to conduct a failure effects analysis on the subassembly level. Work will also continue on mission success and safe abort reliability estimates based upon the

mission time line profiles for a 336-hour LOR mission. Coordination will be assured between NAA/S&ID and Minneapolis-Honeywell on the Apollo preferred parts list. These parts will be incorporated into the flight hardware for the SCS system. Activities are to be undertaken in the performance of stress and parameter variation analyses on SCS circuitry.

Table 4-20. Stabilization and Control Failure Effects Analysis

Item	Drawing Number	Elect. Reference. Desig.	Function	Failure Type	Mission Phase	Failure Effect On Component Performance	Failure Effect On Sub-Systems Performance	Failure Effect On Apollo	Possible Crew Action
Attitude display			Visual display of attitude information	Display incorrect, or input information incorrect			Eliminates accurate attitude information	These displays are very important during critical phases, especially during emergency control of large ΔV. If both displays are inoperative, an abort is suggested and should be successful if the display is only in error.	If only one is incorrect, repair or ignore it. Proceed with available information.
BMAG			Attitude error input, body axes	Short; open; hardover			Eliminates attitude error information if G&N is not functional. If G&N is functional, eliminates IMU back-up for attitude reference	If G&N is not functional, abort has probably been made. If repair can be made, safe return can be accomplished. If hardover occurs during entry, it might not be catastrophic.	If time allows, use G&N reference if available. If G&N is not available, disengage SCS, stabilize with rate and emergency control, and repair. If time does not allow, use best information available; control by emergency.
BMAG			Attitude rate input, body axes	Hardover; open; short			Eliminates back-up to RGP; if RGP is not available, SCS is without-rate information	Effect would depend on conditions of malfunction. If time to repair or go to rate gyros, effect is negligible. During critical phases entry could be catastrophic.	Switch to rate gyros. If not usable, disengage SCS and stabilize displacement display and emergency control. Repair or replace gyro.

Table 4-20. Stabilization and Control Failure Effects Analysis (Cont)

Item	Drawing Number	Elect. Reference. Desig.	Function	Failure Type	Mission Phase	Failure Effect On Component Performance	Failure Effect On Sub-Systems Performance	Failure Effect On Apollo	Possible Crew Action
RGP			Body rates input	Hardover; short to ground			Eliminates rate reference and requires use of BMAG as rate reference	Effect of rate gyro failure depends on conditions. It will be almost impossible to make large ΔV without rate. A successful entry might be made only with one axis inoperative. Safety will depend on redundancy.	Repair of switch to BMAG as rate reference if it is functional.
Accelerometer package			Longitudinal acceleration	Small error; large error positive; large error negative			Higher rate of power consumption due to more usage of G&N and error in trajectory	Higher rate of power consumption due to more usage of G&N and error in trajectory. No need to abort.	Use OFF-ON signal from G&N in subsequent ΔV's. Use manual shut-off. Use G&N in subsequent ΔV's.
SCS control panel			Mode select; deadband adjust; limiter adjust	Switches fail to open or fail to close or short to ground. Proportional adjustments in error or short to ground			Elimination of the control capability, emergency controls must be used until repaired	Abort would be performed. Crew would probably survive.	The mode select panel is redundant. Use redundant control if mode select panel fails. If adjust panel fails, use emergency rate and attitude until faulty control can be isolated.
Gimbal position indicator			Gimbal positioning indication	Open; short; display incorrect			None	None, if automatic control performs gimbal positioning; if automatic fails, could be catastrophic.	None

Table 4-20. Stabilization and Control Failure Effects Analysis (Cont)

Item	Drawing Number	Elect. Reference. Desig.	Function	Failure Type	Mission Phase	Failure Effect On Component Performance	Failure Effect On Sub-Systems Performance	Failure Effect On Apollo	Possible Crew Action
Three-axis rotational control			Attitude command, three axis	Open; short			Use of G&N for extended periods of time. Time in excess of the designed operating time will result in an intolerable reduction of G&N reliability	Degradation of G&N reliability is critical; abort will be made in this case. A successful entry can probably be made providing a permanent G&N failure does not occur.	Switch to G&N attitude control mode.
Delta velocity indicator			Displays; velocity change; provides manual control of SPS	Open; short; incorrect; display			Eliminates the capability of manually performing a ΔV maneuver	Abort would be performed.	Use automatic control, thereby eliminating the need for ΔV indicator.
Electronic control assembly			Signals to SM R/C, CM R/C, and SPS	Short; open			Increased use of fuel; attitude control performance degraded; accuracy of maneuvers are decreased	Abort would probably be made. A successful entry could be made; therefore, the crew would survive.	Use emergency controls.
Display electronics package			Signals to SCS displays	Open; short			Crew is required to use automatic mode, but there is no way to monitor	Abort would be made. A successful entry can be made with automatic control.	Use automatic controls.
Translational Control			Provide ON-OFF signals to reaction control systems	Short; open; control; malfunction			Increased use of G&N	Abort would be performed.	Switch to G&N for automatic translation maneuver.

INSTRUMENTATION

SUMMARY

Equipment of the instrumentation subsystem are being continuously evaluated to determine individual reliability status and the effect of each on over-all Apollo spacecraft reliability. The latest available design information has resulted in analytically determined reliability data shown in Table 4-21.

ANALYSIS

Central Timing Equipment

The redundancy techniques illustrated in Figure 4-47, CTE logic diagram, indicate that the apportioned reliability of 0.9999995 can be achieved. Weight and power requirements can be met through the use of integrated circuits. However, sufficient information to substantiate indicated failure rates may be difficult to obtain. NAA/S&ID Apollo Reliability Engineering is cognizant of this fact and is presently seeking information in the area.

In-Flight Test System

The in-flight test system concept has changed markedly as a result of NASA direction. The present system incorporates 225 individual comparator circuits, allowing that number of measurements to be made. It diagnoses indicated failure to a one- to three-module level. Test points are presently being defined and will be influenced by criticality, accessibility, and maintainability. The manual test set and its associated test panel have been deleted, with an associated decrease in diagnostic capability. The present configuration, however, may allow use of the system voltmeter at other than the 225 hardwired test points. As the result of design information and reliability logic diagrams, the in-flight test system has been apportioned a mission success reliability of 0.99288, with the probability of all 225 comparator tests functioning. A single measurement by the system must not have a probability of failure greater than 0.000215, or a reliability of 0.999785. These figures are based on high-reliability parts and an assumed duty cycle of approximately 10 percent (40 hours).

Table 4-21. Instrumentation System Apportionment and Predictions

Equipment	Predicted Operating Time (hr)	Mission Success Reliability	
		Apportioned	Predicted
Central timing equipment	336	0.9999995	0.99999
In-flight test system	40	0.99288	***
*In-flight test system	40	0.999785	***
**Manual test set			
Television system	40	0.999	0.9993
C-band antenna	10	0.999999	
VHF recovery antenna	72	0.999999	***
VHF recovery antenna backup	72	0.99999	***
HF recovery antenna	72	0.999999	***
2 KMC high gain antenna	336	0.9999	0.99993
Radome	336	0.999999	0.999999
Photographic equipment			
Radiation detection equipment	336	0.99981	***
Entry monitor display	4	****	***

*Single comparator measurement

**Deleted

***Design not available

****In Study

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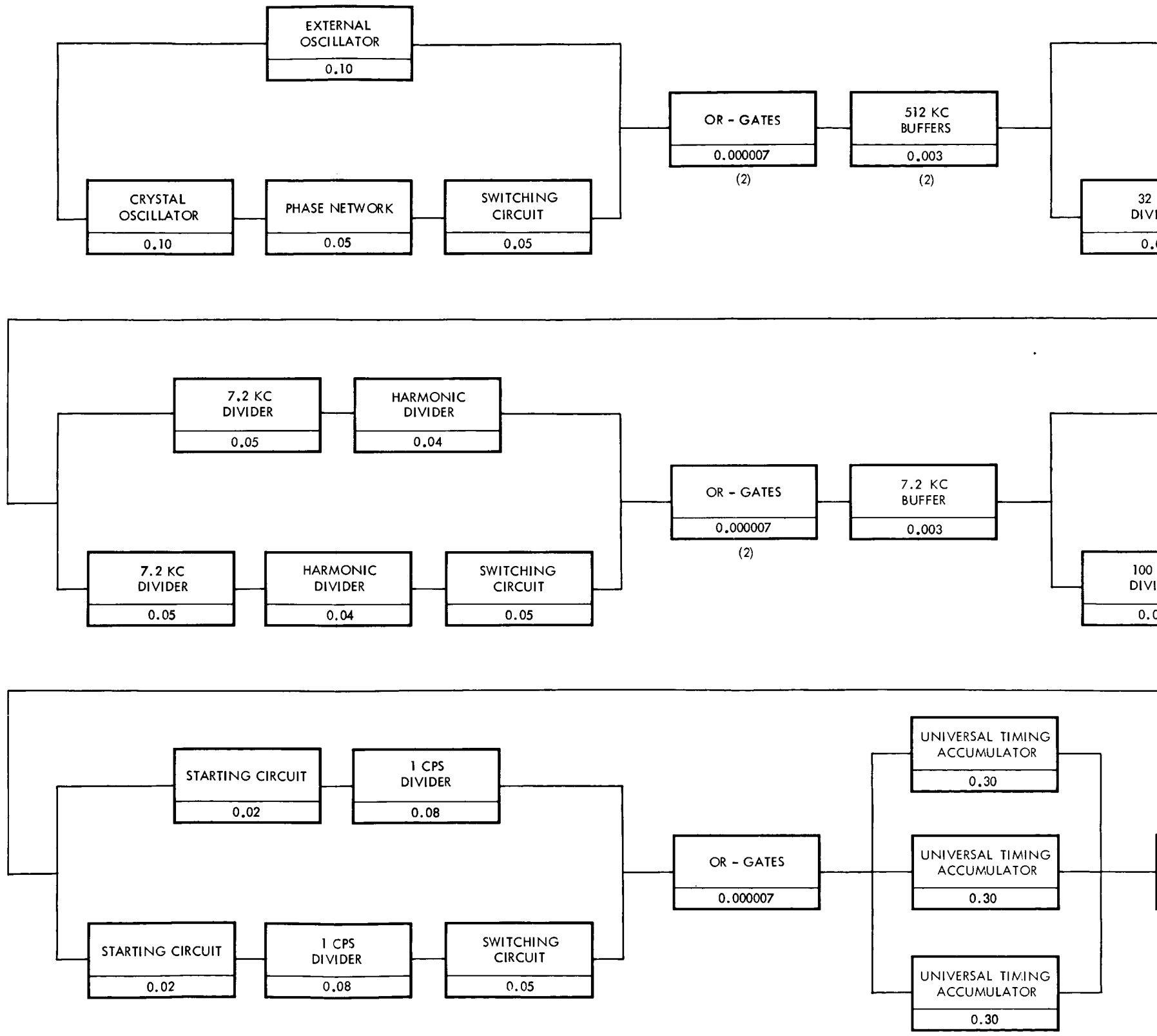
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OR - GATES
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(2)
512 KC BUFFERS
0.003
(2)
32 DIVI
0.
7.2 KC DIVIDER
0.05
HARMONIC DIVIDER
0.04
7.2 KC DIVIDER
0.05
HARMONIC DIVIDER
0.04
SWITCHING CIRCUIT
0.05
OR - GATES
0.000007
(2)
7.2 KC BUFFER
0.003
100 DIVI
0.0
STARTING CIRCUIT
0.02
1 CPS DIVIDER
0.08
STARTING CIRCUIT
0.02
1 CPS DIVIDER
0.08
SWITCHING CIRCUIT
0.05
OR - GATES
0.000007
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UNIVERSAL TIMING ACCUMULATOR
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UNIVERSAL TIMING ACCUMULATOR
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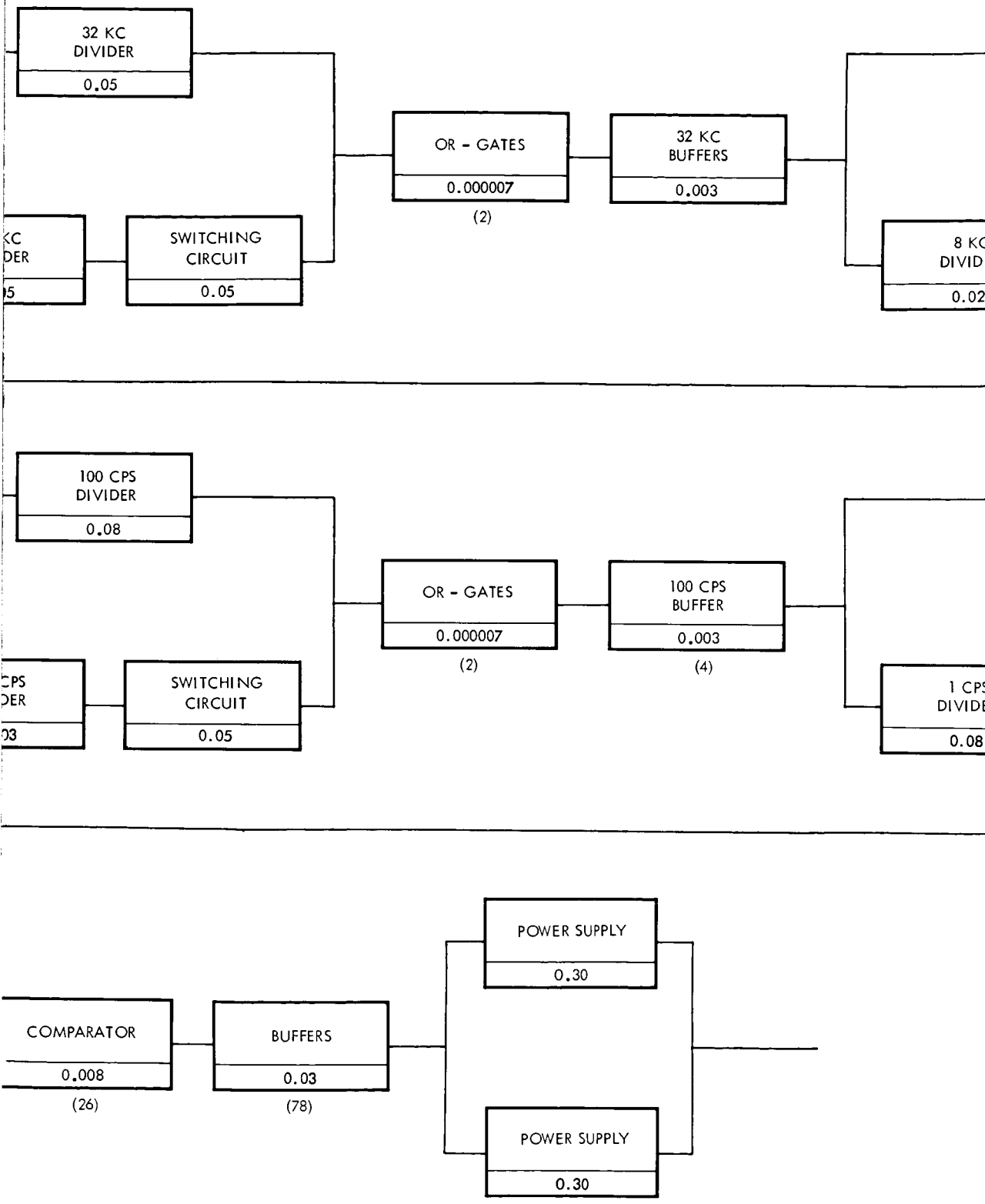
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SWITCHING
CIRCUIT
0.05
OR - GATES
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(2)
32 KC
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100 CPS
DIVIDER
0.08
SWITCHING
CIRCUIT
0.05
OR - GATES
0.000007
(2)
100 CPS
BUFFER
0.003
(4)
COMPARATOR
0.008
(26)
BUFFERS
0.03
(78)
POWER SUPPLY
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POWER SUPPLY
0.30

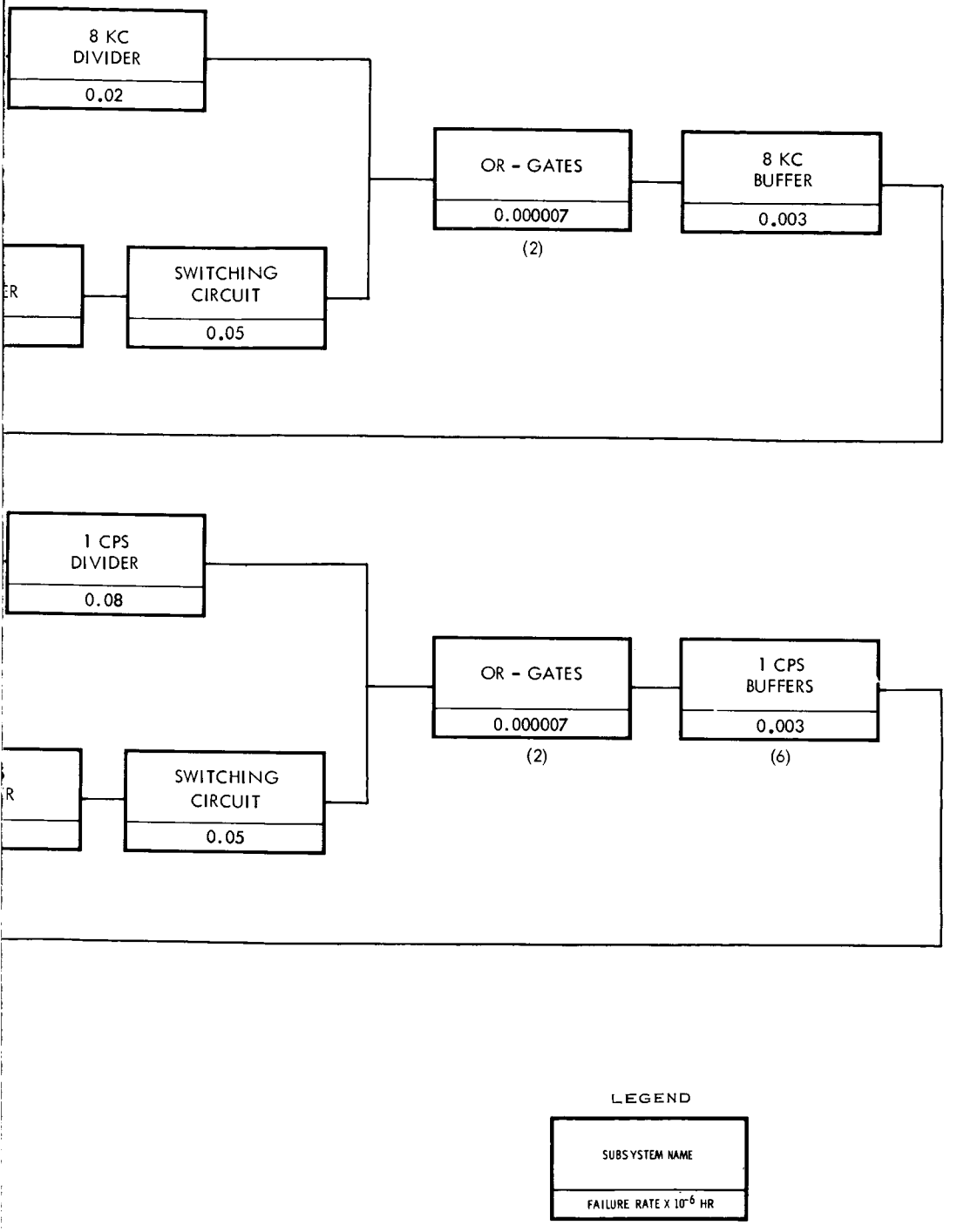

Figure 4-47. Central Timing Equipment Logic Diagram

CONFIDENTIAL

Reliability ramifications of the present in-flight test system configuration are being investigated. Among the considerations are spares determination for over-all reliability, equipment availability, adjustment capabilities, crew availability, and abort causing failures (nonmaintainable).

Equations have been developed, including the effect of the astronauts' ability to perform in-flight maintenance on the probabilities of mission success and crew safety. These equations were submitted to In-Flight Test System Design Engineering for use in setting the design requirements for the in-flight test system. The basic equation for equipment availability is given as

$$E = 1 - (e^{-\mu t}) \cdot (1 - e^{-\lambda T})$$

where

E = Equipment availability

$\mu = 1/\phi$ = Maintenance action rate

ϕ = Mean time to repair - $\frac{\text{total maintenance-action time in hours}}{\text{number of maintenance actions}}$

λ = Failure rate

T = Mission time

t = Maintenance time-constraint

The availability is defined as the probability that a stated percent of the equipment will perform satisfactorily for the time T, with no down time interval exceeding the maintenance time constraint. Thus, equipment availability is a function of reliability and maintainability. Based upon this basic equation, analyses may be performed for evaluation of mission availability, requirements for scheduled maintenance, redundancy factors, and requirements for spares.

In the event of a malfunction indication, the single sensor, illustrated in Figure 4-48 provides a voltage pick-off point to the in-flight test system comparator circuitry that compares the input signal with the reference signal provided by the reference supply. The comparison determines if a signal out-of-tolerance exists. If it does exist, the comparator generates an out-of-tolerance signal. This signal actuates an indicator light advising the crew of the condition. Each comparator has an inhibit gate that blocks or passes an out-of-tolerance signal if the malfunction is determined to be not critical.

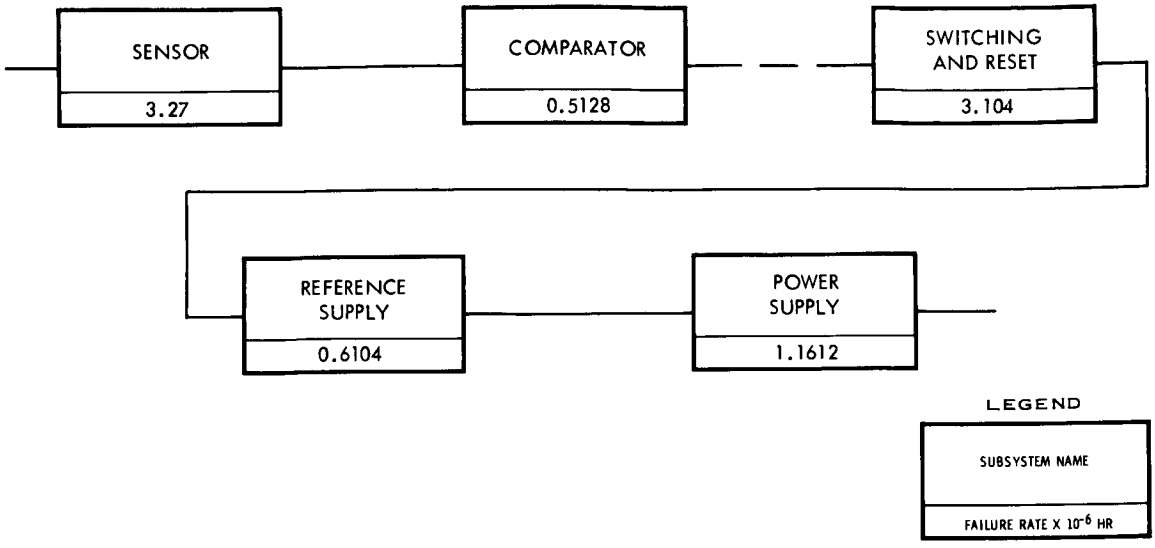

Figure 4-48. Single Sensor and Comparator In-Flight Test System Block Diagram

The IFTS also provides switching and reset circuitry that can transfer any selected analog input or conditioned stimulus signal to the voltmeter for a readout and to the telemetry system for GOSS confirmation. It also can transfer the IFTS stimulus signal into the spacecraft electronic systems to check out systems not actually in use at the time.

The reference and power supplies provide the necessary power for the IFTS.

Television System

The Apollo television camera has been designated nonessential to mission success or crew safety. However, since periodic monitoring of the crew throughout the flight could have ramifications in these reliability areas, Apollo Reliability Engineering has imposed a reliability requirement of 0.999 on this equipment for a minimum of 40 flight operating hours. The selected supplier of the camera, RCA Astro Electronics, has predicted a reliability in excess of this requirement. Several nonstandard components must be evaluated by NAA/S&ID Apollo Reliability Engineering before concurrence is given, however.

Apollo Antennas

Recovery Antenna Equipment

Inputs have been made to the procurement specification for this equipment to ensure contractual obligation of the successful bidder. The inputs include reliability design requirements of 0.999999 for 72-hours minimum operations, environmental conditions for design and packaging, and general quality assurance provisions. It is expected that the contract for this equipment will be let during this quarter, subsequent to proposal evaluations participated in by NAA/S&ID Apollo Reliability Engineering.

2-KMC Hi GH-Gain Antenna

NAA/S&ID Apollo Reliability Engineering has evaluated technical proposals for this equipment from five bidding companies. Each predicted reliabilities in excess of the requirement of 0.9999 for fourteen days of operation. However, none gave sufficient detailed information to substantiate these claims, and these deficiencies were noted as part of the evaluation submitted to Design Engineering. After the contract is awarded, Apollo Reliability Engineering will closely monitor the efforts of the successful bidder for conformance to the reliability requirement.

Radome

Technical proposals from bidding companies were evaluated during this quarter by Apollo Reliability Engineering. Electrical and mechanical reliabilities of 0.999999 and 0.99999999 were established for design requirements. It is recognized, however, that proving the accomplishment of these requirements may be unrealistic and analytical justification difficult.

Radiation Detection Equipment

Sufficient information has been obtained to allow a preliminary apportionment of the radiation detection equipment. The present configuration involves basically two pieces of equipment: the radio detector and the proton detector. Available information indicates that one solar flare is expected during a maximum 14-day lunar mission. Therefore, the probability of a solar flare occurring must be considered 100 percent for a worst-case condition. The radio detector gives approximately 30-seconds warning of a proton shower due to a solar flare. With the reaction control subsystem, the crew orients the spacecraft for maximum protection by mass obstruction. Initial proton encounter is detected by four proton detectors, allowing the crew to determine exact angle of collision and, therefore, optimum orientation of the spacecraft. Each of the equipment

required throughout the foregoing maneuver must be considered essential to crew safety in the event of a solar flare. Assuming equal complexity of the two equipment, apportioned reliabilities have been established at 0.99981 for the radio equipment and 0.99981 for each of the detectors.

Entry Monitor Display

Design information has been insufficient to evaluate realistically the reliability of this equipment, but its function and present operational concept have been analyzed for the reliability logic diagram for reentry. It is used to monitor reentry trajectory by displaying g loading as a function of velocity. It thus provides the astronaut with the information to determine whether the automatic reentry mode is functioning properly, permitting him to guide the command module manually to avoid skip-out or burn-up, if the automatic mode malfunctions. A study is in progress to determine the reliability required of this equipment to satisfy over-all Apollo mission success and crew safety requirements.

Failure Effects Analysis

The results of a preliminary failure effects analysis of the instrumentation subsystem at the equipment level are given in Table 4-26 at the end of this section.

SPECIAL STUDIES

At the request of the NAA/S&ID Electronic Controls and Displays Installation Coordination Committee, Apollo Reliability Engineering has established a list of that electronic equipment that will be affixed to the command module cold plating. Table 4-22 shows the power status of these equipment by mission phase. "Power on" and "Stand by mode" are indicated by (x) and ⓧ, respectively. Table 4-23 shows the related mission phases.

In support of NAA/S&ID Life Systems, Apollo Reliability Engineering is determining how equipment malfunction with all redundancy disregarded will affect the continuity of operations in all mission phases. Critical equipment, as shown on reliability logic diagrams and the mission phase time line, are being combined with Life System Engineering's work load histograms of astronaut activities throughout respective mission phases. Evaluations are being conducted to determine continuity of operations at times when astronauts are 100-percent occupied with activity workloads for maintenance activities. To assure maximum equipment utilization or availability throughout respective mission phases, special consideration is being given to establishing reliabilities with man in the loop by compiling

Table 4-22. Crew Safety Electronic Equipment Mounted to Cold Plate (Revision No. 1)

Electronic Equipment	Mission Phase																								
	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24	25
Sextant				X	X			X		X				X	X					X		X	X		X
Scanning telescope				X	X			X		X				X	X					X		X	X		X
Inertial measuring unit	X	X	X	X	X	X	Ⓧ	X	X	X	X	X	Ⓧ	X	X	X	X	Ⓧ	Ⓧ	X	Ⓧ	X	X	Ⓧ	X
Power and servo assembly	X	X	X	X	X	X	Ⓧ	X	X	X	X	X	Ⓧ	X	X	X	X	Ⓧ	Ⓧ	X	Ⓧ	X	X	Ⓧ	X
Apollo guidance computer	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
Coupling display unit	X	X	X	X	X	X	Ⓧ	X	X	X	X	X	Ⓧ	X	X	X	X	Ⓧ	Ⓧ	X	Ⓧ	X	X	Ⓧ	X
Electronic control assembly							X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
Body-mounted attitude gyro	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
Rate gyro package	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
In-flight test system					X		X						X					X	X		X			X	
DSIF transponder								X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
DSIF power amplifier								X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
PCM telemetry	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
VHF/AM transceiver	X	X	X	X	X	X	X																		
Premodulation	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
Audio center	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
HF transceiver																									
VHF recovery beacon																									
C-band transponder	X	X	X	X	X	X	X																		
Signal conditioning	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
VHF/FM transmitter	X	X	X	X	X	X	X																		
Central timing equipment	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X

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Table 4-22. Crew Safety Electronic Equipment Mounted to Cold Plate (Revision No. 1) (Cont)

Electronic Equipment	Mission Phase																								
	76	77	78	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	95	96	97	98	99	100
Sextant		X	X		X		X	X						X	X		X	X							
Scanning telescope		X	X		X		X	X						X	X		X	X							
Inertial measuring unit	Ⓧ	X	X	Ⓧ	X	Ⓧ	X	X	X	X	X	Ⓧ	Ⓧ	X	X	Ⓧ	X	X	X	X	X	X	X		
Power and servo assembly	Ⓧ	X	X	Ⓧ	X	Ⓧ	X	X	X	X	X	Ⓧ	Ⓧ	X	X	Ⓧ	X	X	X	X	X	X	X		
Apollo guidance computer	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X		
Coupling display unit	Ⓧ	X	X	Ⓧ	X	Ⓧ	X	X	X	X	X	Ⓧ	Ⓧ	X	X	Ⓧ	X	X	X	X	X	X	X		
Electronic control assembly	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X		
Body-mounted attitude gyro	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X		
Rate gyro package	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X		
In-flight test system	X			X		X						X	X			X									
DSIF transponder	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X					
DSIF power amplifier	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X					
PCM telemetry	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X		
VHF/AM transceiver																					X	X	X		
Premodulation processor	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X		
Audio center	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
HF transceiver																									X
VHF recovery beacon																								X	X
C-band transponder																					X	X	X		
Signal conditioning	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X		
VHF/FM transmitter																					X	X	X		
Central timing equipment	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X		

Table 4-22. Crew Safety Electronic Equipment Mounted to Cold Plate (Revision No. 1) (Cont)

Electronic Equipment	Mission Phase																								
	51	52	53	54	55	56	57	58	59	60	61	62	63	64	65	66	67	68	69	70	71	72	73	74	75
Sextant				Ⓧ	Ⓧ		Ⓧ		Ⓧ		Ⓧ	Ⓧ		X	X					X	X				
Scanning telescope				X	X		X		X		X	X		X	X					X	X				
Inertial measuring unit	Ⓧ	Ⓧ	Ⓧ	X	X	Ⓧ	X	Ⓧ	X	Ⓧ	X	X	X	X	X	X	X	X	Ⓧ	X	X	X	X	X	Ⓧ
Power and servo assembly	Ⓧ	Ⓧ	Ⓧ	X	X	Ⓧ	X	Ⓧ	X	Ⓧ	X	X	X	X	X	X	X	X	Ⓧ	X	X	X	X	X	Ⓧ
Apollo guidance computer	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
Coupling display unit	Ⓧ	Ⓧ	Ⓧ	X	X	Ⓧ	X	Ⓧ	X	Ⓧ	X	X	X	X	X	X	X	X	Ⓧ	X	X	X	X	X	Ⓧ
Electronic control assembly	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
Body-mounted attitude gyro	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
Rate gyro package	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
In-flight test system	X	X	X			X		X		X								X							X
DSIF transponder	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
DSIF power amplifier	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
PCM telemetry	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
VHF/AM transceiver	X	X	X																						
Premodulation	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X		X	X
Audio center	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X		X	X
HF transceiver																									
VHF recovery beacon																									
C-band transponder																									
Signal conditioning	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
VHF/FM transmitter																									
Central timing equipment	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X

Table 4-22. Crew Safety Electronic Equipment Mounted to Cold Plate (Revision No. 1) (Cont)

Electronic Equipment	Mission Phase																							
	26	27	28	29	30	31	32	33	34	35	36	37	38	39	40	41	42	43	44	45	46	47	48	49	50
Sextant	X					X		X	X					Ⓧ	Ⓧ									Ⓧ	
Scanning telescope	X					X		X	X					X	X									X	
Inertial measuring unit	X	X	X	Ⓧ	Ⓧ	X	Ⓧ	X	X	X	X	Ⓧ	X	X	X	Ⓧ	Ⓧ	Ⓧ	Ⓧ	Ⓧ	Ⓧ	Ⓧ	Ⓧ	X	Ⓧ
Power and servo assembly	X	X	X	Ⓧ	Ⓧ	X	Ⓧ	X	X	X	X	Ⓧ	X	X	X	Ⓧ	Ⓧ	Ⓧ	Ⓧ	Ⓧ	Ⓧ	Ⓧ	Ⓧ	X	Ⓧ
Apollo guidance computer	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
Coupling display unit	X	X	X	Ⓧ	Ⓧ	X	Ⓧ	X	X	X	X	Ⓧ	X	X	X	Ⓧ	Ⓧ	Ⓧ	Ⓧ	Ⓧ	Ⓧ	Ⓧ	Ⓧ	X	Ⓧ
Electronic control assembly	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
Body-mounted attitude gyro	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
Rate gyro package	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
In-flight test system				X	X		X					X				X	X	X	X	X	X	X	X		X
DSIF transponder	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
DSIF power amplifier	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
PCM telemetry	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
VHF/AM transceiver																	X	X	X	X	X	X	X	X	X
Premodulation	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
Audio center	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
HF transceiver																									
VHF recovery beacon																									
C-band transponder																									
Signal conditioning	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
VHF/FM transmitter																									
Central timing equipment*	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X

maintenance action rates, maintenance time constraints, and basic equipment reliabilities. A set of reliability logic diagrams as related to in-flight tests and maintenance functions for continuity of operation requirements has been constructed for the respective phases of the LOR mission. These diagrams are shown in Figure 4-49.

Table 4-23. Lunar Orbital Rendezvous Mission Phases for Cold-Plate Study

Phase No.	Phase Description
1	Launch to 100,000 feet
2	100,000 feet to LET jettison
3	LET jettison through earth orbit injection
4	Earth orbit, IMU alignment
5	Earth orbit, Nav sighting
6	Earth orbit through translunar injection
7	Translunar coast cisearth (period 1)
8	LEM transfer
9	First midcourse correction (translunar), IMU alignment
10	First midcourse correction (translunar), nav sighting
11	First midcourse correction (translunar), attitude maneuver
12	First midcourse correction (translunar), ΔV
13	Translunar coast period 2
14	Second midcourse correction (translunar), IMU alignment
15	Second midcourse correction (translunar), Nav sighting
16	Second midcourse correction (translunar), attitude maneuver
17	Second midcourse correction (translunar), ΔV
18	Second midcourse correction (translunar), attitude hold
19	Translunar coast period 3
20	Nav sighting (translunar)
21	Translunar coast period 4
22	Nav sighting, IMU alignment (translunar)
23	Nav sighting (translunar)
24	Translunar coast period 5
25	Third midcourse correction (translunar), IMU alignment
26	Third midcourse correction (translunar), nav sighting
27	Third midcourse correction (translunar), attitude maneuver
28	Third midcourse correction (translunar), ΔV
29	Third midcourse correction (translunar), attitude hold
30	Translunar coast period 6
31	Nav sighting (translunar)
32	Translunar coast period 7

Table 4-23. Lunar Orbital Rendezvous Mission Phases for Cold-Plate Study (cont)

Phase No.	Phase Description
33	Fourth midcourse correction (translunar), IMU alignment
34	Fourth midcourse correction, nav sighting
35	Fourth midcourse correction (translunar), attitude maneuver
36	Fourth midcourse correction (translunar), ΔV
37	Fourth midcourse correction (translunar), attitude hold
38	Lunar orbit injection at 80 nautical miles
39	Lunar orbit nav period 1, IMU align
40	Lunar orbit nav period 1, nav sighting
41	Lunar orbit coast
42	Crew tasks and check out
43	Injection into equal period orbit
44	Transfer to 50,000 feet
45	Equal period orbit stay
46	Perilune retro (50,000 to 1,000 feet)
47	Hover, translation and touchdown
48	LEM lunar surface stay period 1
49	C/M nav sighting
50	LEM lunar surface stay period 2
51	Lunar ascent to circular orbit at 50,000 feet
52	Ascent to 80 nautical miles
53	Rendezvous and docking
54	Lunar orbit nav, IMU alignment
55	Lunar orbit nav, nav sightings
56	Lunar orbit coast
57	Lunar orbit nav sightings
58	Lunar orbit coast
59	Lunar orbit nav sightings
60	Lunar orbit coast
61	Lunar orbit nav, period 3, IMU alignment
62	Lunar orbit nav, period 3, nav sightings
63	Transearth injection
64	First midcourse correction (transearth), IMU alignment
65	First midcourse correction (transearth), nav sightings
66	First midcourse correction (transearth), attitude maneuver
67	First midcourse correction (transearth), ΔV, large
68	First midcourse correction (transearth), ΔV, small
69	Transearth coast period 1
70	Second midcourse correction (transearth), IMU alignment

Table 4-23. Lunar Orbital Rendezvous Mission Phases for Cold-Plate Study (cont)

Phase No.	Phase Description
71	Second midcourse correction (transearth), nav sighting
72	Second midcourse correction (transearth), attitude maneuver
73	Second midcourse correction (transearth), ΔV, large
74	Second midcourse correction (transearth), ΔV, small
75	Second midcourse correction (transearth), attitude hold
76	Transearth coast period 2
77	Nav sightings (transearth), IMU alignment
78	Nav sightings (transearth)
79	Transearth coast period 3
80	Nav sightings (transearth)
81	Transearth coast period 4
82	Third midcourse correction (transearth), IMU alignment
83	Third midcourse correction (transearth), nav sighting
84	Third midcourse correction (transearth), attitude maneuver
85	Third midcourse correction (transearth), ΔV, large
86	Third midcourse correction (transearth), ΔV, small
87	Third midcourse correction (transearth), attitude hold
88	Transearth coast period 5
89	Transearth nav sightings, IMU alignment
90	Transearth nav sightings
91	Transearth coast period 6
92	Fourth midcourse correction (transearth), IMU alignment
93	Fourth midcourse correction (transearth), nav sightings
94	Fourth midcourse correction (transearth), attitude maneuver
95	Fourth midcourse correction (transearth), ΔV, large
96	Fourth midcourse correction (transearth), ΔV, small
97	S/M separation
98	Entry
99	Droque chute deployment through landing
100	Post-landing

TEST PROGRAM

Qualification Test

The following information on the instrumentation system was submitted for incorporation into Ground Qualification Tests, Vol. III, SID 62-109 (General Test Plan Research and Development for Project Apollo Spacecraft).

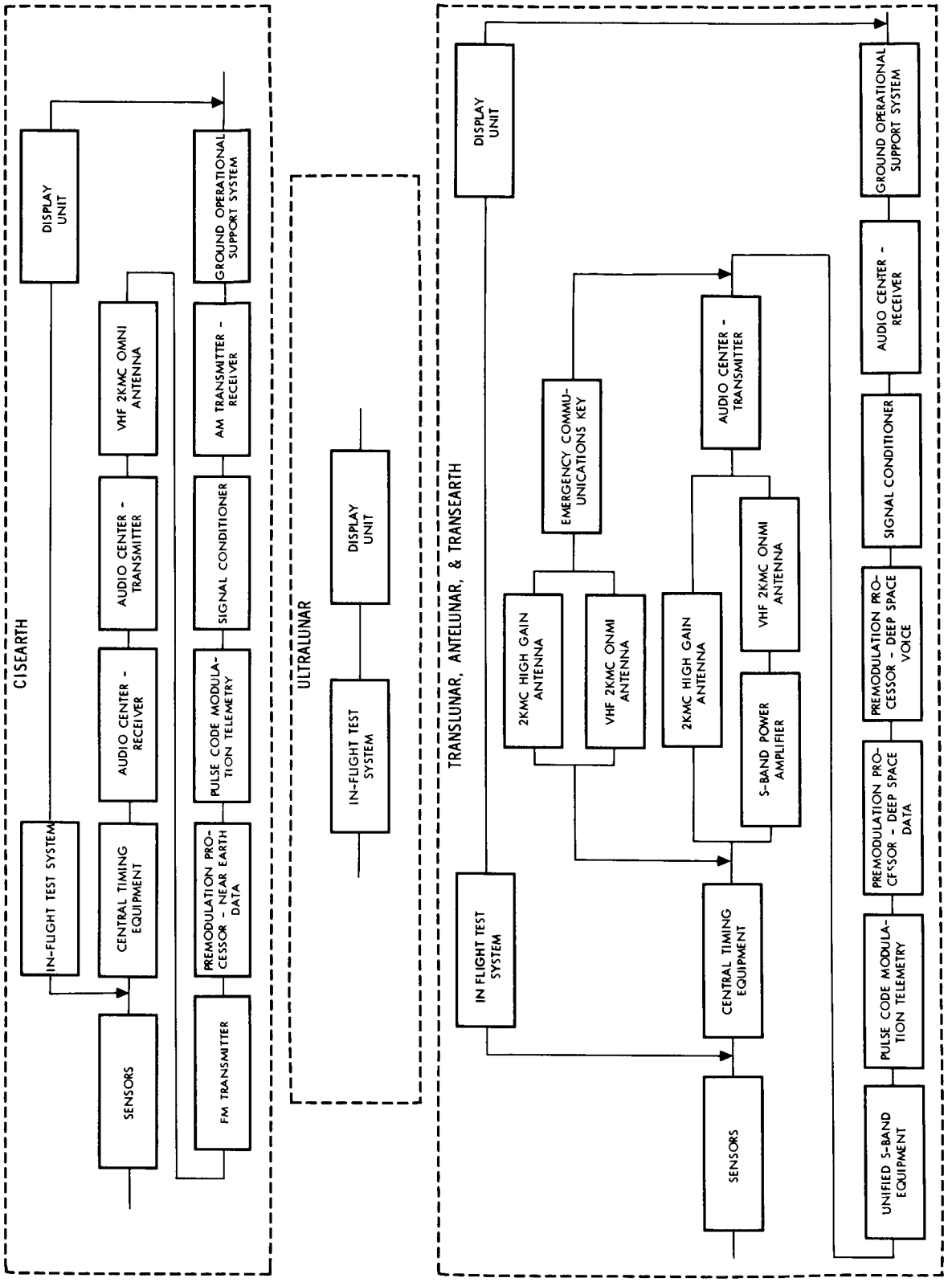

Figure 4-49. Monitor Logic for Sparing

1. The central timing equipment test oriented reliability logic diagram

2. The criticality and hardware requirements for each component of the instrumentation system

3. The qualification test schedule

The present status of the Apollo antenna equipments is given in Table 4-24.

Table 4-24. Apollo Antenna Equipment Status

Equipment	Bid Release	Supplier Selected	Due Tests	Quality Tests	Accept Tests	Percent Completion
Recovery antenna	x	x				40
High gain antenna	x					20
Radome antenna	x					20
R&D beacon antenna	x	x	x	x		80
R&D telemetry antenna	x	x	x			60
Beacon antenna	x	x				40

Test Schedules

Table 4-25 is an anticipated milestone chart for the receiving of development, qualification, and acceptance test data for the Apollo antenna equipments.

Table 4-25. Apollo Antenna Equipment Anticipated Milestone

Equipment	Development	Qualification	Acceptance
Recovery antenna	5-1-63	7-8-63	11-1-63
High-gain antenna	9-1-63	1-15-64	3-15-64
Radome antenna	9-13-63	1-16-64	3-15-64
Beacon antenna	7-15-63	1-15-64	3-15-64
R&D beacon antenna	11-2-63	11-30-62	4-1-63
R&D telemetry antenna	12-26-62	3-1-62	3-8-63

SUBCONTRACTOR MANAGEMENT

Subcontractor Coordination

Elgin National Watch Company has been selected to supply the Apollo central timing equipment. Apollo Reliability Engineering conducted a field survey to determine, in detail, the basis of the Elgin cost proposal for this equipment. Subsequently, a cost negotiation base was generated by Apollo Reliability Engineering in support of Purchasing for each reliability task described by Elgin. Support will be given Purchasing at cost negotiations during the next reporting period.

RCA has been selected as the supplier for the Apollo television equipment. NAA/S&ID Apollo Design and Reliability Engineering conducted a presentation at RCA for the purpose of defining the procurement specification.

A bidders conference concerning the high-gain antenna system was held in January 1963 to inform suppliers of S&ID's latest design and reliability requirements.

PLANNED ACTIVITIES

Following is an outline of instrumentation subsystem activities planned for the second quarter of 1963.

- Failure effects analyses
  - Equipment used on AFRM 011 (Subassembly level)
- Subcontractor documentation review
  - Reliability predictions
  - Stress analysis
  - Parameter variation analysis
  - Failure mode and effects analysis
  - Standard and nonstandard parts list
  - Failure reporting and analysis program
  - Design reviews
  - Monthly progress reports
- In-house analyses
  - Equipment reliability status
  - Parts evaluations
  - In-flight test system
    - Test points required
    - Ramifications of present configuration
  - Sensor application
    - Redundancy required
    - Measurements required
    - Reliability requirements
  - Cold plate priority
    - Electronic equipment power status by mission phase
    - Criticality
- Proposal evaluations
    - In-flight test system
    - Recovery antenna equipment
  - Procurement specifications inputs
    - Design reliability requirement
    - Environmental criteria
  - Subcontractor management
    - Coordination meetings
    - Parts selection
    - Failure effects analyses
    - Parameter variation analyses
    - Failure reporting and analyses

Table 4-26. Failure Effects Analysis

Item	Drawing Number	Elect. Reference. Desig.	Function	Failure Type	Mission Phase	Failure Effect On Component Performance	Failure Effect On Sub-Systems Performance	Failure Effect On Apollo	Possible Crew Action
In-flight test system			Monitors spacecraft operations, revealing system or subsystem malfunctions automatically	Open or Circuit a) Programmer b) Comparator c) Storage d) Control Panel e) Display Panel	All phases	Loss	None	None, unless one of the following: FM transmitter, signal conditioner, pulse code modulation, telemetry, sensors, or central timing equipment has failed, then abort (near earth)	Self-test mode and manual test set allow diagnosis for maintenance.
Central timing equipment			Provides primary mode of timing for various electronic equipments; provides accumulated time and time to event displays; correlates all data as an integral part of time—critical operations	Failed SYNC input from G&N plus failed internal oscillator; open or shorted outputs	All phases	Loss	Loss of the in-flight test system, the television, and the displays.	Abort if not corrected.	Repair or adjust if possible.
Television equipment			Provides ground monitoring of the crew	Open or shorted output; failed vidicon	All phases	Loss	None	None	Spare vidicon provided. No other compensation.
Radiation detection equipment			Monitors radiation levels interior and exterior to the CM; readouts provided for crew as well as telemetry	Open or shorted output	All phases	Loss	None	Possible crew radiation exposure excessive.	None
C-band antenna			C-band radar tracking antenna providing tracking up to 4000 miles from earth	Open or shorted input to the antenna	Near earth	Loss	None	Abort prior to the translunar injection. If one of the following has failed: CTE, sensors, signal conditioner, PCM telemetry, FMX, P-P near earth, flight director rate indicator.	None

Table 4-26. Failure Effects Analysis (Cont)

Item	Drawing Number	Elect. Reference. Desig.	Function	Failure Type	Mission Phase	Failure Effect On Component Performance	Failure Effect On Sub-Systems Performance	Failure Effect On Apollo	Possible Crew Action
2KMC high-gain antenna			Cislunar communication antenna	Open or shorted input to the antenna	Deep-space	Loss	None	None, unless S-band power amp failed then abort.	None
VHF recovery antenna equipment (2-antennas)			Meets the antenna requirements for VHF recovery beacon which provides line-of-sight direction finding capabilities	Open or shorted input to the antenna; broken radiating element.	Parachute deployment through pickup	Loss	None	Degraded recovery capabilities. HF transceiver and antenna remain as recovery aids.	None
HF recovery antenna			Meets the antenna requirements for the HF transceiver for directional finding and communications during recovery	Open or shorted input to the antenna; deployment mechanism failure; broken radiator.	Recovery	Loss	None	Degraded recovery capabilities. VHF recovery equipment remains.	None
Sensors			Monitor various physical parameters and environmental conditions of the spacecraft and its systems. Information is provided to IFTS and/or is telemetered to Earth. Critical measurements are displayed directly to the crew	Structural failure; signal conditioner; open or shorted output	All phases	Loss	None	Possible crew and ground ignorance of failed critical system.	None
Entry monitor display			Display velocity versus acceleration and lift to provide a manual back to automatic reentry	Accelerometer failure	Entry	Loss	None	None, unless automatic reentry made fails, then probable loss of crew.	None

V. GROUND SUPPORT EQUIPMENT ANALYSIS

SUMMARY

A comprehensive reliability program has been established for Apollo ground support equipment (GSE) to assure maximum equipment effectiveness. This will be done by evolving systems that have high inherent characteristics of reliability, availability, and performance. The following design requirements are used as guides to achieve the desired effectiveness:

1. Allowable frequency of critical failure modes that affect spacecraft inherent in-flight mission success probability

2. Allowable frequency of critical failure modes which affect spacecraft inherent in-flight crew safety probability

3. Mean-time-between-failures (including nuisance as well as catastrophic failures).

Those GSE equipments that are used in a closed-loop final checkout and during the countdown portion of the prelaunch operation and that could affect the success of the mission or crew safety are assigned to a mission essential equipment category. The comprehensive reliability program previously mentioned is applied to those equipments.

The effort expended to date in apportioning requirements and analyzing equipments to determine compliance with these requirements is presented herein. Information is also provided that includes an interpretation of work statement requirements and how these requirements are translated into equipment design goals, a brief description of the methods of design analysis used, a preliminary definition of the Apollo prelaunch operational readiness program, a definition and status of analysis of mission essential equipments, and a delineation of planned future activities.

OPERATIONAL READINESS

Work Statement Requirements

The basic criterion for GSE is the probability that the GSE will not cause the spacecraft reliability to fall below reliability requirement 0.9638 and shall be at least 0.999. The crew safety requirements are correspondingly 0.9995 for spacecraft and 0.9999 for GSE. This is interpreted to mean

that while the GSE is operating the probability of the GSE creating or inducing a failure in the spacecraft shall be no greater than 0.001 (i.e., one failure/ 10^3 missions), when considering mission success, and 0.0001 (i.e., one failure/ 10^4 missions), when considering crew safety. The following studies are being undertaken to insure the achievement of these apportioned reliabilities.

1. Failure mode and effect analyses include consideration of the following types of failures and their subsequent effect on spacecraft subsystems.

   Probability of accepting bad spacecraft equipment (β error)

   Probability of rejecting good spacecraft equipment (α error)

   Probability of inducing failures into spacecraft systems that are subsequently detectable or undetectable.

   Probability of GSE failures occurring in such a manner that subsequent spacecraft system failures could not be detected. In the case of open failures, GSE indicators might indicate that the GSE is operating properly.

2. Probability analyses are performed subsequent to the failure mode and effects analyses. These analyses are conducted as a means of determining that the GSE complex is compliant with its work statement requirement. If compliance is not achieved, these analyses will allow pinpointing of specific areas where redesign can be considered as a means of effecting the required levels of improvement.

Apollo Prelaunch Operational Readiness Program Requirements

This program is designed to insure launch capability of the spacecraft within the required launch envelope (i.e., in phase with the moon's corridor window) by utilizing the results of operational studies that define this final operation in terms of usage cycles, failure modes, frequency of failures, and task times (i.e., times required to perform checkout or to conduct those unscheduled maintenance jobs that result from equipment failures).

Equipment mean-time-between-failure (MTBF) requirements are established for all mission essential equipments used during the count-down portion of the prelaunch operation for the Apollo. These MTBF's are used as programmatic inputs into a computer program that simulates the ground operating cycle during prelaunch operations. These MTBF requirements are established at such levels that with including the frequency of occurrence of failures, considered in combination with task times (both scheduled and

unscheduled) and logistic supply (spares, numbers of GSE, manpower, etc.), the requirements can be fulfilled within a predesignated elapsed period of time. The program outputs of the computer simulation includes Apollo operational readiness, percent of waiting time for all logistics, and statistical distributions about a mean for all these factors, including MTBF's.

Translated Equipment Design Goals

Each mission essential GSE is apportioned the following reliability requirements.

1. Allowable frequency of critical failure modes that affect spacecraft inherent mission success probability

2. Allowable frequency of critical failure modes that affect spacecraft inherent crew safety probability

3. MTBF's (including nuisance as well as catastrophic failures)

Requirements 1 and 2 are apportioned as a means of achieving the previously discussed work statement requirements. Requirement 3 is apportioned in compliance with the time restrictions of the prelaunch period.

CLASSIFICATION OF EQUIPMENTS

The five general types of GSE (i.e., auxiliary, checkout, handling, servicing, and training) are subdivided into three categories depending on their intended usage. The categories are divided as follows:

1. Mission Essential Equipment. Those systems that are directly involved in a closed-loop operation with spacecraft systems and the function of which may have an effect on mission success or crew safety.

2. Operating Ground Equipment (OGE). Those equipments utilized in the support of mission essential equipments, spacecraft systems, checkout area, staging areas, manufacturing facilities, etc.

3. Support Ground Equipment (SGE). Those equipments and facilities utilized in transportation, handling, maintenance, recovery, shelter, etc.

Table 5-1 presents a listing of all presently defined mission essential equipments. As previously mentioned, only the mission essential equipments contribute to the required mission success and crew safety probabilities, and these are apportioned accordingly. MTBF requirements are established for

Table 5-1. GSE List of Mission Essential Equipment

Model No.	Title
A14-003	Pyrotechnics Initiators Substitute Unit
A14-019	Disconnect Set, Umbilical
A14-024	Disconnect Set, Umbilical
C14-002	Barometric Switch Test Unit
C14-011	CM and SM RCS Checkout Group
C14-019	Test Conductor Console
C14-051	Pyrotechnics Bench Maintenance Equipment
C14-075	Propulsion System Checkout Group
C14-112	Radar Transponder and Recovery Beacon Checkout Unit
C14-177	Cable Set Electrical
C14-180	Cable Set Electrical
C14-200	Carry-On Command System Decoder
C14-201	Carry-On Command Unit
C14-210	Carry-On PCM System
C14-211	Carry-On Signal Conditioning and Sample Unit No. 1
C14-212	Carry-On Signal Conditioning and Sample Unit No. 2
C14-213	Carry-On Signal Conditioning and Sample Unit No. 3
C14-214	Carry-On Signal Conditioning and Sample Unit No. 4
C14-215	Carry-On Signal Conditioning and Sample Unit No. 5
C14-230	Data Interleaving and Command System
C14-231	External Digital Test Command Unit
C14-240	Servicing Equipment Pace Adapter

Table 5-1. GSE List of Mission Essential Equipment (Continued)

Model No.	Title
C14-260	Line Matching Amplifier
S14-001	Module Leak Test Unit
S14-002	N_2O_2 Transfer Unit
S14-005	Water Transfer Unit
S14-007	Pressurization Contamination Prevention Unit
S14-008	50-50 UDMH and N_2H_4 Transfer Unit
S14-009	Helium Transfer Unit
S14-018	Freon Transfer Unit
S14-019	Ethylene Glycol Water Transfer Unit
S14-026	LH_2 Transfer Unit
S14-032	LO_2 Transfer Unit
S14-041	Fluid Distribution System
S14-042	Fluid Distribution Unit
S14-052	Water-Glycol Cooling Unit
S14-053	Ethylene Glycol Fluid Trim Control
S14-054	Fuel Cell Power Plant Water-Glycol Servicing Unit
S14-057	Reaction Control System Oxidizer Servicing Unit
S14-063	Service Module Reaction Control System Fuel Servicing Unit
S14-064	Command Module Reaction Control System Fuel Servicing Unit

OGE and SGE. The purpose of this effort is to minimize the frequency of unavailability of these types of equipments consistent with the economics associated with the program.

ANALYSIS

The following analyses are performed on electrical, electro-mechanical, and mechanical circuits to ensure proper design consideration is given to all elements contributing to mission success of crew survival.

1 A computer-aided Mandex d-c analysis is performed on all electronic circuitry to determine stress conditions and failures that can occur because of poor design conditions. The stress analysis is used to calculate a predicted reliability number in terms of failure rate by applying the stress parameters obtained to derating factors and failure rate application.

2 Failure mode and effect analysis studies are performed on all mission essential equipment to determine critical failure modes that would result in a situation that GSE would induce failures into the spacecraft or result in the GSE equipment failing to detect a spacecraft failure. As a result of these studies, reliability makes recommendations to the design group. Table 5-2, Failure Mode and Effect Analysis - GSE, defines, in general terms, criticality, corrective action, and failure mode classification.

Summary of Analyses Performed

The following analyses have been performed on various GSE to date. A summary of the completed analyses follows.

1 Reliability predictions

Model No.	Title	Minimum MTBF Reliability Requirement	Predicted MTBF
S14-001	Module Leak Test	600	1575
S14-002 & S14-008	Fluid Transfer Unit	600	3745
S14-002	Helium Booster Unit	600	2922
S14-026	LH_2 Fluid Transfer Unit	600	1470
C14-009	Crew System Checkout Unit	600	4591
A14-003	Pyrotechnics Substitute Unit	600	2400

Table 5-2. Failure Mode and Effect Analysis - GSE Criticality, Corrective Action, and Failure Mode Classification

Criticality Categories	Failure Mode Order	Corrective Action Classification (CAC)
Catastrophic	1st order failure mode: Failure of a GSE component that results in personnel hazard or a spacecraft failure, detected or undetected.	1. Unit will be out of service, and possibly removed from work area, while failed component is being replaced and cause of failure determined.
Major	2nd order failure mode: Failure of a GSE component that results in temporary delay of servicing or checkout function.	2. Unit will be out of service while minor replacement or repairs are being made. It will not be necessary to remove unit from work area.
Minor	3rd order failure mode: Failure of a GSE component that results in little or no delay of servicing or checkout function.	3. Minor adjustments will be made while the servicing or checkout function continues.
All others	4th order failure mode: Failure of a GSE component that results in no delay in servicing or checkout function.	4. No corrective action required until servicing or checkout function in completed.

2. Failure mode and effect analysis, LH_2 transfer unit S14-002

3. Mandex worst-case analysis pyrotechnics substitute unit A14-003.

As previously stated, each piece of GSE must show compliance with three requirements. Therefore, the equipment in the preceding list have MTBF's in excess of their requirements and are not deemed acceptable until all analyses are completed. Hence, all analyses completed to date must be considered preliminary only. The remainder of this section presents detailed analyses of the LH-2 transfer system and pyrotechnic simulator. Results of analyses of the remainder of the units in the list will be presented in the next quarterly issue of this report.

FAILURE MODE AND EFFECT ANALYSIS (LH_2 TRANSFER UNIT-S14-026)

Summary

This report presents the results of a failure mode effect analysis performed on the liquid hydrogen transfer unit. The analysis covers the operational usage of the unit, conditioning of the sc storage vessels, filling the storage vessels, and supplying the fuel cell with fuel.

Table 5-3, Failure Modes Effect Analysis Summary, illustrates the results of the analysis performed and indicates the serious undetectable first-order failure due to contamination joining the media stored in the service module.

Table 5-4 shows the tabulated analysis on the S14-026 LH_2 transfer unit. Definitions of entries on the Failure Mode Effect Analysis are also included.

In performing the analysis it was necessary to make the following assumptions.

1. All control valves open or close upon receiving a signal from the remote control station. The signal activates a solenoid that controls a pneumatic cylinder which opens or closes the valve gate. Functional failure of the valve operations could cause a system failure.

2. Leaks in joints, connections, or plumbing accessories could cause a system failure.

3. Permitting contamination to join the media stored in the spacecraft vessel could cause a first-order spacecraft failure.

Table 5-3. Failure Mode Effect Analysis Summary

Effect	Failure Mode			Corrective Action Classification			Delay Servicing Operation	Halt Servicing Operation	Halt Servicing, Damaged Components, Personnel Hazard
	I	II	III	1	2	3			
Undetectable malfunction during a transfer - GSE and personnel hazard	5	2		4	3		5		2
May cause personnel hazard	2				1	1	2		
May cause undetectable failure	2	5	1	1	5	2	5	3	
May damage interface systems	2					2	2		
May cause detectable failure	8	2	3	2	3	8	6	4	3
Totals	19	9	4	7	12	13	20	7	5
Total failure modes: 32									

Table 5-4. Failure Mode Effect Analysis - GSE

Function Phase	Subsystem Component or Item	Failure Mode	Failure Effect On			Remarks
			System	Other Systems or Subsystems	Function	
GN_2 purge	Valve system	Fail open	Cannot shut off	Depletes GN_2 stores	Delays transfer operations	2nd order failure - corrective action classification (CAC) No. 2 (Repairs made during a transfer mode create a personnel hazard since particles contamination could enter the SC storage system.
		Fail closed			Halts servicing operation	3rd order failure - CAC No. 3 (Repairs create a system hazard that may cause undetectable failure.)
	Filter system	Fractures, ruptures, fails to filter	Permits contamination to bypass filter system	Contaminates SC storage system	Delays	1st order failure - CAC No. 1 (Contamination of SC storage vessels)

Table 5-4. Failure Mode Effect Analysis — GSE (Continued)

Function Phase	Subsystem Component or Item	Failure Mode	Failure Effect On			Remarks
			System	Other Systems or Subsystems	Function	
GN_2 purge (Continued)		Fails closed (clogged)	Blocks flow		Halts	1st order failure - CAC No. 1 (Detectable)
	Gage and sensor system	Fail to report, reports inaccuracies	Cannot accurately monitor system		Delays transfer operation	2nd order failure - CAC No. 2 (Repairs hazardous to system integrity – Undetectable)
GH_2 conditioning system	Valve system	Fail open	Cannot shut off	Depletes GN_2 store	Delays transfer operation	2nd order failure - CAC No. 2 (Detectable)
		Fail closed			Halts servicing operation	3rd order failure - CAC No. 3 (Detectable)
	Plumbing system	Leaks	Escaping gaseous hydrogen	Depletes stores	Delays transfer	1st order failure - CAC No. 1 (Personnel hazard – may fail to detect)
	Filter system	Ruptures, fails to filter	Permits contaminents to bypass filter system	Contamnates SC stores	Delays transfer operation	1st order failure - CAC No. 1 (Personnel hazard – may fail to detect)

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Table 5-4. Failure Mode Effect Analysis – GSE (Continued)

Function Phase	Subsystem Component or Item	Failure Mode	Failure Effect On			Remarks
			System	Other Systems or Subsystems	Function	
GH_2 conditioning system (Continued)		Clogs	Shutoff flow, slow flow blocks flow		Delays transfer operation	2nd order failure - CAC No. 3 (Detectable)
	Gage and sensor system	Fail to register, registers inaccuracies	Cannot accurately monitor system		Delays transfer operations	2nd order failure - CAC No. 2 (May cause undetectable failure)
LH_2 fluid transfer	Valve system	Fail open	Cannot shutoff	Creates hazard to SC	Delays transfer	1st order failure - CAC No. 3 (May damage interface)
		Fail close		Permits mobile storage to deteriorate	Delays transfer	3rd order failure - CAC No. 3 (Detectable)
	Pump system	Fails to operate	No fluid flow	SC storage vessels lose conditioning.	Halts servicing operation	1st order failure - CAC No. 3 (Detectable)

Table 5-4. Failure Mode Effect Analysis - GSE (Continued)

Function Phase	Subsystem Component or Item	Failure Mode	Failure Effect On			Remarks
			System	Other Systems or Subsystems	Function	
LH_2 fluid transfer (Continued)	Pump impeller	Fractures	Fire or explosion	All	Halts servicing operation	1st order failure - CAC No. 3 (May damage interface systems)
	Gage and sensor system	No readout	Cannot monitor operation		Halts servicing operation	1st order failure - CAC NO. 2 (Detectable)
		False readout	Cannot control operation	SC storage operation	Inaccur-ate transfer, delays operation	2nd order failure - CAC No. 2 (Undetectable)
Valves 17	Hand shutoff 3	Fail open	Loss of control over supply source	Supply source	Delays operation	1st order failure - CAC No. 2 (Damage to GSE)
		Fail closed				1st order failure - CAC No. 3 (Appreciable delay)

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Table 5-4. Failure Mode Effect Analysis – GSE (Continued)

Function Phase	Subsystem Component or Item	Failure Mode	Failure Effect On			Remarks
			System	Other Systems or Subsystems	Function	
Valves 17 (Continued)		Leak	Media flow out of control	Depletes stores		2nd order failure mode - CAC No. 2 (Damage to equipment – personnel hazard)
	Regulator 3	Adjust-ment fails to regulate	Flow out of control		Delays operation	2nd order failure mode - CAC No. 2
		Leak to ambient	Loss of pressure		Delays operation	2nd order failure mode - CAC No. 2 (Undetectable)
	Solenoid 5	Fail open	Loss of control over the fluid distribu-tion system	Hazard-ous to propel-lant man-agement system	Delays operation	1st order failure - CAC No. 2 (Personnel hazard)
		Fail closed	Loss of control over system		Delays operation	1st order failure - CAC No. 3 (Appreciable delay – personnel hazard)

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Table 5-4. Failure Mode Effect Analysis – GSE (Continued)

Function Phase	Subsystem Component or Item	Failure Mode	Failure Effect On			Remarks
			System	Other Systems or Subsystems	Function	
Valves 17 (Continued)	Check and pressure relief 6	Fail closed	Pressurization amplification damages components.		Halts servicing operation, personnel hazard	1st order failure mode - CAC No. 2 (May cause damage to stage)
Plumbing attenuated attachments 143	Flexible hoses, disconnects, joints and couplings	Leak or burst	Loss of media	Propulsion management system and GSE hazard	Halts servicing	1st order failure mode - CAC No. 1 (Personnel and GSE hazard)
Electrical units	Connections	Break or deteriorate thru corrosion	Loss of operating control		Halts servicing	1st order failure mode - CAC No. 1 (Undetectable)
GH_2 fuel	Valve system	Fail closed	None	Fuel cell	Halts operation	1st order failure mode - CAC No. 1 (Interrupts CM power during the definitive time envelope period)

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Table 5-4. Failure Mode Effect Analysis – GSE (Continued)

Function Phase	Subsystem Component or Item	Failure Mode	Failure Effect On			Remarks
			System	Other Systems or Subsystems	Function	
GN_2 fuel cell purge	Valve system	Fail closed	None	Fuel cell	Halts operation	3rd order failure mode - CAC No. 3 (Appreciable delay during prelaunch preparations)
Main Power 110 volts, 60 cycle ac, and 28 volts dc	Relays (to be identified later)	Fails open	Loss of main	Loss of 28 volts control	Halts servicing operation	1st order failure - CAC No. 3 (Detectable)
	Relays (to be identified later)	Fail closed			Delays trouble shooting	1st order failure - CAC No. 3 (Undetectable)
	Pump motor	Fails to operate	No fluid flow, no pressure		Halts servicing operation	1st order failure - CAC No. 3 (Detectable)
	Other					

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4. Sensory system failure could permit transmitting erroneous information to the control panel.

5. Failure in the venting system could cause pump cavitation and interrupt the transfer, causing a delay.

6. Heat leakage in the fluid transfer system would permit the LH_2 to boil, producing a stream of bubbles that would cause vapor lock or pump cavitation and a system failure. Positioning of relief valves and adequate subcooling facility would be a restraint opposing this failure mode.

Intended Usage

The LH_2 transfer unit will be used to control the flow of the cryogen fuel from the mobile storage unit (S14-066) to the spacecraft storage vessels during filling and draining operations. Figure 5-1, Suggested Schematic of Suppliers LH_2 Transfer Unit S14-026, illustrates the operation of the system, parts, vents, and components, as defined herein. The LH_2 transfer unit will also control gaseous hydrogen to the fuel cell during its prelaunch operational period. Prior to fueling the spacecraft, the LH_2 transfer unit will control the purging of gaseous nitrogen to the spacecraft storage vessels and the spacecraft fuel cell. Following the purge cycle, the unit will control conditioning the spacecraft storage vessels with gaseous hydrogen to prepare the vessels for cryogenic storage.

All these functions will be performed in such a manner as to prevent contaminating the spacecraft stored cryogenic fuel. The transfer unit includes a filtering system capable of removing all particles over 5 microns in size and 98 percent of all particles over 2 microns in size from the media leaving the outlet ports. However, particles entering downstream from the transfer unit will contaminate the spacecraft storage system. In the event contamination presents itself downstream, GSE Design has considered the possibility of recirculating the media as a method of removing contamination. This involves returning the contaminated media from the spacecraft via the vent distribution system to the LH_2 transfer unit through the filter system, to be recirculated to the spacecraft through the fluid distribution system.

GSE Design has considered a suggestion to include a boil-off tank located in the tower to receive the LH_2. This tank, maintained at 1 atmosphere, would be used to permit the cryogens to cool down before entering spacecraft. The resultant gases would be passed to vent and the liquid fuel gravity fed into the spacecraft vessels.

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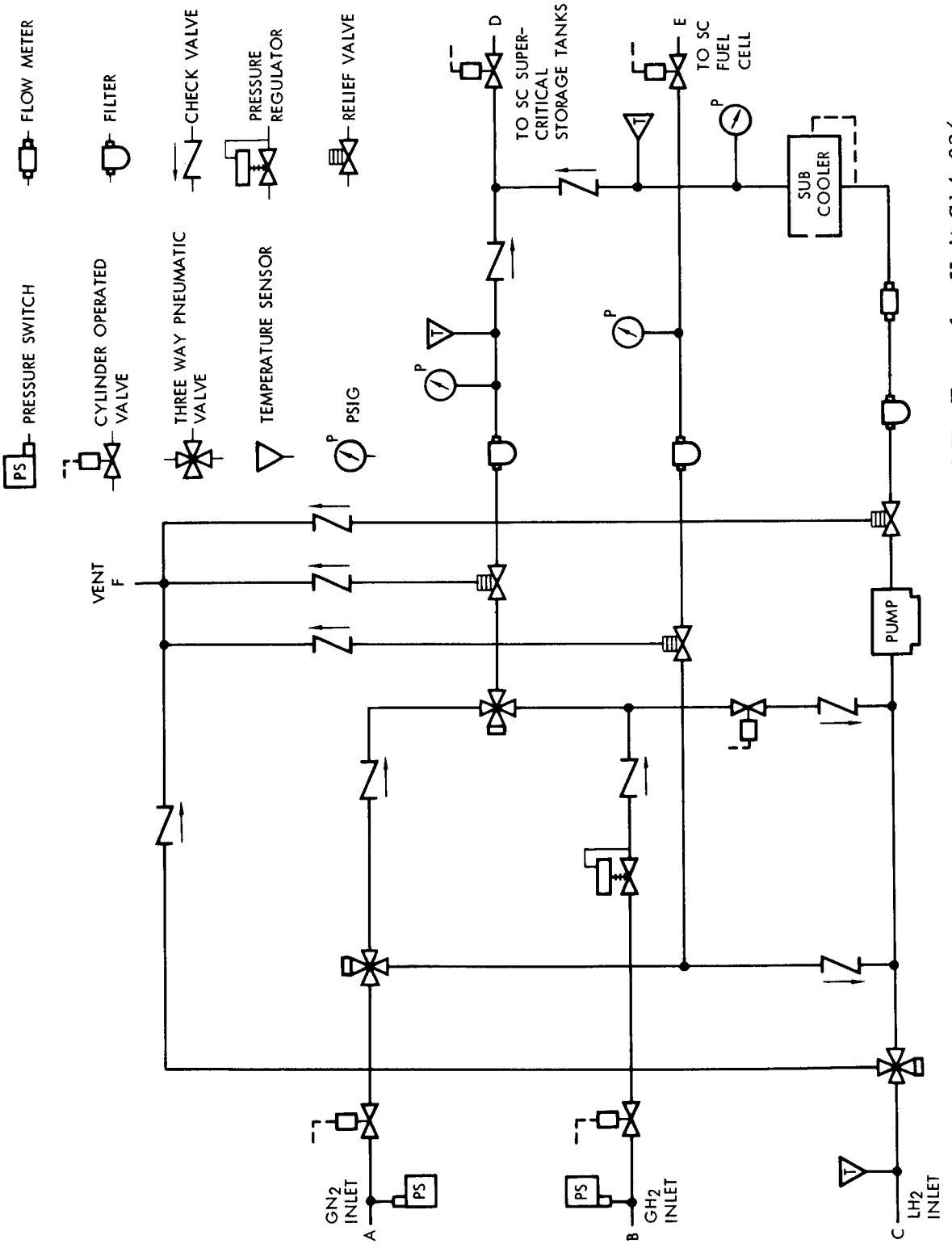

Figure 5-1. Suggested Schematic of Suppliers LH_2 Transfer Unit S14-026

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Functional Description

The LH_2 transfer unit is a collection of pumps, valves, subcoolers, control consoles, plumbing components, electrical connections, and other accessories grouped into a system. The unit's fundamental function is:

1. To fill the spacecraft LH_2 storage vessels to capacity with the fluid media within the specified period of the time envelope

2. To control the GN_2 and GH_2 flow of gases through the LH_2 spacecraft storage system

3. To control GN_2 to the fuel cell for purging

4. To control GH_2 to the fuel cell for operational support during countdown

5. To drain the stored media from the spacecraft vessels

The unit is operated from a control panel located on the console or from a similiar control panel remotely located. The station having control prevents the dead station energizing through an interlock circuit. The unit is powered by connecting it to the 115V 60 ~ input line. The unit will be in continuous operation during the final 240 minutes of the prelaunch time envelope.

The pneumatic control valves are constantly pressurized by gaseous nitrogen bled from the GN_2 system through the regulator valve. The remote control station (RS) monitors the control valves through 28V d-c circuits to the solenoids. All control valves will fail-safe by assuming their normal position following a power or pneumatic failure.

The stage propellant management system will make and break the spacecraft quick disconnections on signals from RS. The system will leak check all disconnects during the transfer operation.

The GN_2 distribution system, the GH_2 distribution system, and the LH_2 transfer system all vent through check valves to the vent system. A failure of any unit of the vent system would be a serious first-order failure.

The unit is instrumented to provide visual indications to determine the stress status the mediae exert on the system. The stresses sensed are pressure, temperature, and acceleration.

Conclusions and Recommendations

The order of failure referred to in the analysis sheets under the "remarks" column indicates the number of events that must occur simultaneously in order to create the "Failure Effect" on the function indicated.

At the present time the design of the unit is too premature to fully analyze the failure modes of the circuitry that will be designed into this unit. However, as the design progresses, a failure mode effect analysis of the circuitry will be made and incorporated into this report.

The recommendations listed below have been submitted to the design group for their evaluation.

1. No recirculating media for the purpose of removing contaminants. (The relatively small quantities involved suggest it would be economically feasible to pass a contaminated media through the vent system to the facility disposal.)

   Insure a make-up redundancy of media on hand in the LH_2 Mobile Storage Dewar (S14-066).

   Use a subcooler device in the tower at spacecraft level, vented through the vent system.

   Pass tower subcooler outlet through a filtering system.

   Test for contamination by sampling downstream from the tower filtering device.

   Provide a bypass connection to divert a contaminated media to vent.

2. Provide the Remote Control Station and the unit console with instrumentation capable of monitering the spacecraft system stresses. The stresses are temperature, pressure, and volumetric capacity.

Work Sheets Explanation

A sample Failure Mode Effect Analysis worksheet is shown in Table 5-4. An explanation of the worksheet is as follows:

1. Function Phase. A specific operational or time element of an over-all GSE function

2. Subsystem, Component, or Item. Items listed include only those that are active in a particular function phase (part most likely to fail)

3. Failure Mode. The manner in which an equipment malfunctions (e.g., "short" or "open" circuit for electrical malfunctions and "closed" or "open" for mechanical valves, etc.) (most likely mode of failure)

4. Failure Effect on System. The effect of a specific malfunction on the primary GSE system of which the malfunctioning unit is a part (effect of failure on system)

5. Other Systems or Subsystems. Failure effect of a specific malfunction on other related GSE systems or subsystems (effect of failure on subsystems)

6. Failure Effect on Function. The effect of a specific malfunction on the GSE system functional requirements, as related to the prime equipment being serviced and/or supported (effect of failure on function)

7. Remarks. Order of failure summary of event leading to failure and supplementary statements qualifying or further defining failure effect.

MANDEX WORST-CASE (D-C ANALYSIS) PYROTECHNICS SUBSTITUTE UNIT - A14-003

Introduction

This portion of the report presents the results of a reliability study on the pyrotechnic initiators substitute unit (A14-003). Data are presented on the system, subsystem, and part levels, including the results of the Mandex Worst-Case d-c computer analysis that was performed on each electronic circuit in the unit. Examination of the Mandex results revealed that no subcircuit failures were incurred due to worst-case variations in the subcircuit parameters and that all component worst-case power dissipation levels were well within tolerance extremes. A detailed failure mode and effects analysis is not included here, but a subsequent account will be reported at a later date.

The analysis is composed of three phases. The first contains a tabular review of the Mandex worst-case d-c analysis results for the various subcircuits composing the explosive bridge wire simulator and hot-wire or squib simulator. The second presents a probability analysis utilizing actual operational power stresses and temperatures to predict the failure rates of the subcircuits on a parts level. The third gives the reliability failure rate predictions for the system subcircuits utilizing parts generic failure rates.

Mandex worst-case d-c analyses were performed to evaluate subcircuit ability to indicate the mandatory voltage and current levels for accurate simulator operation under part and parameter tolerance variations. Parameter variation computer programs were also run on the subcircuits, and these printouts, along with the Mandex, are available for review within the Computer Technology Section of Apollo Reliability.

The failure rate data presented were obtained from the AVCO and Martin Reliability Handbooks. An evaluation of the system and subsystem failure rates is thus made, based on the best state-of-the-art data currently available from accurate sources of in-service data. Predicted on the failure rate data compiled from the evaluation of the unit's subcircuits, the failure rate and MTBF of the pyrotechnic initiator substitute unit are 41.972 percent per 1000 hours and 2400 hours, respectively. The calculated MTBF compares favorably with the minimum required MTBF of 600 hours.

A failure mode and effect analysis, utilizing computer-aided techniques, will be performed and reported subsequently. This analysis will determine first- and second-order failure modes or failures of the simulator components that result in personnel hazard, detected or undetected spacecraft failure, and the probability of accepting bad spacecraft equipment or rejecting good spacecraft equipment during prelaunch checkout. Failure inducing shorts and opens on the part, subsystem, and system level can be simulated readily in an IBM 7090 computer program, yielding results indicative of the initiator simulator performance under these conditions and the failure effects produced with respect to the spacecraft.

A brief discussion of the pyrotechnic initiator simulator unit's usage and functional operation is presented in the following paragraphs.

Intended Usage

The pyrotechnic initiator simulator set (A14-003) will be utilized to simulate the pyrotechnic initiators by interfacing with the pyrotechnic firing system. It will exhibit the proper initiator impedances, measure the outputs of the system, and transmit signals indicating the adequacy of the measurements to GSE.

The pyrotechnic initiators on board the spacecraft are explosive devices responsible for activating various systems by intense concentrations of heat, gas generation, or shock wave from detonation. There are two types of pyrotechnic initiators. The hot-wire or squib initiator utilized a low-voltage d-c current to initiate combustion of an explosive charge. The explosive bridge wire initiator utilizes a high-energy pulse across a spark gap to initiate the explosive charge. The firing of these initiators at predetermined times is the end function of the pyrotechnic system.

Composing the pyrotechnic initiator simulator set are groups of squib and explosive bridge wire (EBW) simulators. The purpose of the initiator simulators is to simulate the squib and EBW initiators during sequencer checkout and provide output signals indicating the performance of the spacecraft pyrotechnic firing unit during checkout.

Functional Description

Explosive Bridge Wire Simulator

The output of the EBW firing unit consists essentially of a capacitor discharged into a load. The EBW simulator presents a resistant load to the firing unit. The voltage applied across the discharge resistor is reduced by a voltage divider, fine adjusted by a potentiometer, and fed into the input of the comparator circuit. Here it is compared with the output of the reference element to determine whether the discharge voltage is greater than the reference voltage at a specified time. The existence of such a condition is sufficient to cause the initiation of an explosive charge by the firing unit. At this point the output of the comparator becomes true. It is fed simultaneously to the control input of the pulse "AND" gate and the input of the one-shot multivibrator, which changes state for a time duration determined by an external capacitance. The true-going "false" output of the one-shot is fed to the set input of a flipflop, which becomes true and causes the transient output indicator to become true. This indicates that the simulator has received a pulse from the firing unit. The normally true output of the one-shot is fed into the pulse input of the pulse "AND" gate. As this input again becomes true, while the control input is true, the output of the pulse "AND" gate becomes true, causing the GO-NO-GO flipflop to set. The GO output then becomes true and the NO-GO false, indicating that the simulator has received a satisfactory pulse. The existence of one-shot and comparator simultaneously true outputs fulfills the condition of the discharge voltage being greater than a certain value at a specified time and insures that firing of the initiators will be accomplished. A block diagram of the EBW simulator unit is illustrated in Figure 5-2.

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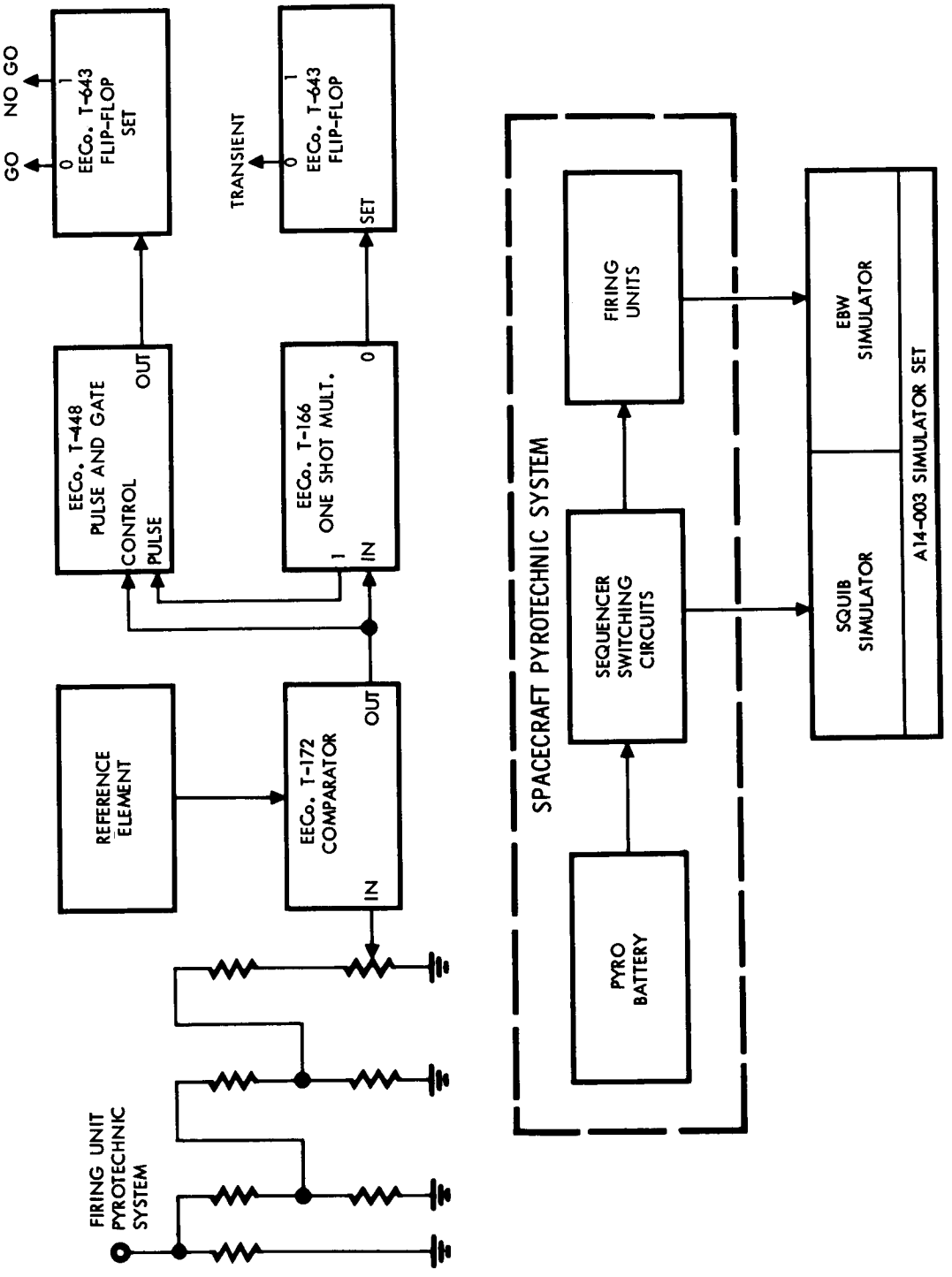

Figure 5-2. EBW Simulator Block Diagram

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Squib Simulator

The squib simulator presents to the firing unit a circuit breaker resistance equal to the resistance of the actual hot-wire initiator. Upon reception of an adequate voltage from the firing unit through auxiliary contacts of a relay, the circuit breaker closes, initiating a GO signal through its contacts. The firing voltage is propagated through an input resistor and diode to the gate lead of a forward biased silicon control rectifier (SCR). This is sufficient to cause the SCR to conduct. The signal transmitted to the gate of a second SCR is delayed by an resistor-capacitor network for a time duration equal to the closing time of the circuit breaker. After the circuit breaker is sufficiently closed, the second SCR will fire, activating relay K1 and opening its auxiliary input contacts, cutting off the input current and resetting the circuit breaker. The SCR units will continue in the forward conducting state until reset by an external source that reverse-biases the units below their breakdown voltage.

Mandex Worst-Case d-c Analysis Summary

Mandex computer analyses were performed on all the subcircuits of the pyrotechnic initiator simulator unit to determine if the subcircuits could provide the outputs necessary for reliable simulator operation when subjected to induced stresses of worst-case part and parameter values. The results of the analysis follow.

Subcircuit operational states were determined and the appropriate equivalent circuits constructed. Nodes were chosen throughout each circuit, and the operational matrix equations were written and programmed for computer worst-case evaluation.

Table 5-5 exhibits the nominal output values, along with the worst-case maximum and minimum computed values and the maximum and minimum allowable values. Differences in the allowable and worst-case computer output values have also been documented. Examination of the results indicate that all subcircuit worst-case output voltages remained well within their maximum and minimum allowable variations. Part and subsystem worst-case power stresses were also shown to be well within tolerance when subjected to parameter variation.

Probability of Failure Subcircuit Parts Analysis

The subcircuit parts for the pyrotechnic initiator simulator system were analyzed for probability of failure. The failure rate data utilized in the analysis was obtained from the AVCO Reliability Engineering failure rate curves. In compiling the over-all subcircuit failure rate on the part level, an application factor (KA) was employed in every case. This factor

Table 5-5. Mandex Worst-Case Output Results

Output Parameters	Nominal Solutions	Worst Case Maximum	Max Allowed Worst Case	Difference	Worst Case Minimum	Min Allowed Worst Case	Difference
DUAL PULSE "AND" GATE T-448							
V_1	- 3.022	- 2.652	- 2.500	0.152	-3.415	- 3.5	0.085
V_2	- 7.114	- 6.052	- 5.700	0.352	-8.147	- 8.500	0.352
V_3	- 5.455	- 4.643	- 4.400	0.243	-6.337	- 6.700	0.362
V_4	- .018	- 0.017	- 0.015	0.002	-0.059	- 0.06	0.003
V_5	- 1.330	- 1.089	- 1.000	0.089	-1.599	- 1.600	0.000
V_6	-10.687	- 9.496	- 9.400	0.096	-11.880	-12.000	0.120
ONE-SHOT MULTIVIBRATOR T-166							
V_1	-3.37	-2.79	-2.00	0.079	-4.05	-4.2	0.15
V_2	-4.18	-3.51	-2.00	1.51	-4.95	-5.0	0.05
V_3	-3.76	-3.04	-2.5	0.54	-4.57	-4.7	0.13
V_4	-4.38	-3.57	-3.2	0.37	-5.29	-5.4	0.11
V_5	-3.63	-3.02	-2.7	0.32	-4.33	-4.5	0.17
V_6	-4.61	-3.87	-3.3	0.57	-5.40	-5.43	0.03

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Table 5-5. Mandex Worst-Case Output Results (Continued)

Output Parameters	Nominal Solutions	Worst Case Maximum	Max Allowed Worst Case	Difference	Worst Case Minimum	Min Allowed Worst Case	Difference
ONE-SHOT MULTIVIBRATOR T-166							
V_7	-4.11	-3.46	-3.2	0.26	-4.83	-5.0	0.17
V_8	-3.38	-2.80	-2.1	0.70	-4.05	-4.1	0.05
V_9	-1.78	-1.42	-10.0	0.42	-2.27	-2.8	0.53
V_{10}	-11.2	-10.0	-9.6	0.40	-12.42	-12.7	0.28
RESET SET TOGGLE FLIP FLOP T-643							
V_1	-11.56	-11.41	-10.00	1.410	-11.705	-13.00	1.295
V_2	-11.43	-11.25	-10.50	0.749	-11.602	-12.00	0.398
V_3	-11.56	-11.33	-11.00	0.334	-11.770	-12.50	0.727
V_4	-11.47	-11.24	-10.50	0.739	-11.695	-12.50	0.806
V_5	- 1.75	- 1.47	- 1.00	0.468	- 2.03	- 2.40	0.374
V_6	- 4.23	- 3.89	- 3.70	0.193	- 4.59	- 5.00	0.411
V_7	- 4.23	- 3.89	- 3.70	0.194	- 4.59	- 5.00	0.411
V_8	- 4.24	- 3.90	- 3.70	0.199	- 4.59	- 5.00	0.406

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Table 5-5. Mandex Worst-Case Output Results (Continued)

Output Parameters	Nominal Solutions	Worst Case Maximum	Max Allowed Worst case	Difference	Worst Case Minimum	Min Allowed Worst Case	Difference
RESET SET TOGGLE FLIP FLOP T-643							
V_9	- 4.24	- 3.90	- 3.70	0.198	- 4.59	- 5.00	0.407
V_{10}	- 4.24	- 3.90	- 3.70	0.198	- 4.59	- 5.00	0.407
V_{11}	- 3.29	- 3.01	- 2.50	0.511	- 3.58	- 4.00	0.420
V_{12}	- 3.95	- 3.57	- 2.70	0.874	- 4.33	- 5.00	0.670
V_{13}	- 3.62	- 3.28	- 2.70	0.578	- 3.97	- 4.00	0.332
V_{14}	- 3.29	- 3.01	- 2.70	0.314	- 3.58	- 4.00	0.416
V_{15}	- 2.99	- 2.65	- 2.50	0.146	- 3.34	- 3.50	0.159
V_{16}	-11.73	-11.59	-11.00	0.594	-11.86	-12.50	0.637
SQUIB SIMULATOR							
V_1	20.66	22.64	24.00	1.36	18.87	16.00	2.87
V_2	20.08	22.06	24.00	1.94	18.28	16.00	2.21
V_3	27.15	28.72	30.00	1.28	25.58	22.00	3.58
V_4	21.79	23.74	25.00	1.26	19.81	18.00	1.81

Table 5-5. Mandex Worst-Case Output Results (Continued)

Output Parameters	Nominal Solutions	Worst Case Maximum	Max Allowed Worst Case	Difference	Worst Case Minimum	Min Allowed Worst Case	Difference
SQUIB SIMULATOR							
V_5	-4.99	-2.71	0.00	2.71	-7.31	-8.00	0.685
V_6	-5.62	-3.43	0.00	3.43	-7.83	-8.00	0.169
V_7	3.37	4.14	6.00	1.86	2.67	2.00	0.670
V_8	0.980	1.21	3.00	1.79	0.765	0.00	0.765
VOLTAGE COMPARATOR T-172							
V_1	3.75	4.27	4.5	0.22	3.11	3.00	0.11
V_2	3.95	4.37	4.75	0.37	3.49	3.51	0.02
V_3	-11.78	-10.7	-10.3	0.45	13.19	-13.50	0.309
V_4	-10.07	-8.71	-8.7	0.012	-11.47	-11.50	0.021
V_5	-9.38	-8.20	-7.48	0.22	-10.4	-10.64	0.24
V_6	-2.76	-2.27	-2.0	0.27	-3.30	-4.0	0.69
V_7	-5.92	-4.92	-4.50	0.42	-6.43	-6.7	0.27

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was based on the actual part operational temperature and power stresses and was used to modify the generic failure rate to yield a rate commensurate with the failure rate encountered in actual system operation.

Each circuit of the initiator simulator set was analyzed, and the analysis presented in Tables 5-6 and 5-7. The tables contain the stress ratios of the component parts, the generic failure rates, the application factors, and the total failure rates predicted for each circuit on the basis of these values.

The tables yield a failure rate for the squib simulator of 0.655 percent per 1000 hours and an MTBF of 15,300 hours. The EBW simulator failure rate and MTBF are 5.055 percent per 1000 hours and 19,780 hours, respectively.

Initiator Simulator Failure Rate Analysis

The failure rates for the initiator simulator were analyzed. The generic failure rate data used in the analysis was obtained from the Martin Handbook of component parts failure rates compiled from actual military parts applications. The analysis is presented in Table 5-8. The failure rates for the subcircuits were compiled by evaluating the reliability of the circuits on the component part level. The results of this method of failure rate determination will be used as a basis of comparison for the AVCO operational failure rate analysis for verification of values and accuracy determination.

The analysis yields a failure rate for the squib simulator of 0.3443 percent per 1000 hours. The total failure rate for the EBW simulator is 5.268 percent per 1000 hours, while the MTBF is 19,000 hours.

[redacted]

Table 5-6. Probability Analysis - Stress Ratio

VOLTAGE COMPARATOR T-172					
				Max Rate at 80 C	
Part No. and Description	Quantity	Mfg Recommended Maximum Use	Actual Max Use (Milliwatts)	Ratio	Absolute (Milliwatts)
Q_1 2N 1478	1	Power = 250 mw	0.005	0.00002	0.005
Q_2 2N 1478	1	BVCEW = 20 volts	41.0	0.164	41.0
Q_3 2N 1304	1	Power = 150 mw	0.003	0.00002	0.003
Q_4 2N 1304	1	BVCEO = 20 volts	25.2	0.168	25.2
R_1	1	125 mw	0.011	0.00008	0.011
R_2	1	125 mw	36.7	0.294	36.7
R_3	1	125 mw	3.78	0.029	3.78
R_4	1	125 mw	12.6	0.101	12.6
R_5	1	125 mw	2.43	0.019	2.43
R_6	1	125 mw	17.5	0.142	17.5
R_7	1	125 mw	0.013	0.001	0.013
R_8	1	125 mw	4.76	0.038	4.76
R_9	1	125 mw	11.84	0.096	11.84
R_{10}	1	125 mw	8.79	0.070	8.79

[redacted]

Table 5-6. ProbabilityAnalysis - Stress Ratio (Continued)

DUAL PULSE "AND" GATE T-448					
Part No. and Description	Quantity	Mfg Recommended Maximum Use	Actual Max Use (Milliwatts)	Max Rate at 80 C	
				Ratio	Absolute (Milliwatts)
Q_1 2N438	1	Power = 150 mw BVCEO = -25 volts	0.019	0.00014	0.021
Q_2 2N428	1	Power = 150 mw BVCEO = -12 volts	0.032	0.00021	0.03
CR_1 1N3125	1	80 mw BV = -20 volts	0.001	0.000013	0.001
R_1	1	Power = 125 mw	3.52	0.028	3.5
R_2	1	Power = 125 mw	23.8	0.190	23.7
R_3	1	Power = 125 mw	0.526	0.0042	0.525
R_4	1	Power = 125 mw	9.38	0.075	9.47
R_9	1	Power = 125 mw	0.002	0.000016	0.002
R_{11}	1	Power = 125 mw	5.05	0.04	5.0
R_{12}	1	Power = 125 mw	7.51	0.06	7.5
R_{13}	1	Power = 125 mw	19.8	0.158	19.5
R_{14}	1	Power = 125 mw	1.18	0.009	1.2
R_{15}	1	Power = 125 mw	4.21	0.033	4.15
C_1, C_3, C_5, C_6	4			0.06	

R = Resistor
Q = Transister
C = Capacitors
CR = Diodes

Table 5-6. Probability Analysis - Stress Ratio (Continued)

ONE-SHOT MULTIVIBRATOR T - 166					
				Max Rate at 80 C	
Part No. and Description	Quantity	Mfg Recommended Maximum Use	Actual Max Use (Milliwatts)	Ratio	Absolute (Milliwatts)
Q_1 2N438	1	Power = 150 mw BVCEO = -25 volts	19.6	0.131	19.6
Q_2 2N438	1	Power = 150 mw BVCEO = -25 volts	16.35	0.109	16.35
Q_3 2N438	1	Power = 15 volts BVCEO = -25 volts	1.2	0.08	1.2
R_1	1	Power = 125 mw	1.94	0.0155	1.94
R_2	1	Power = 125 mw	2.37	0.019	2.37
R_3	1	Power = 125 mw	2.14	0.0171	2.14
R_4	1	Power = 125 mw	2.6	0.021	2.6
R_5, R_{14}	2	Power = 125 mw	3.68	0.029	3.68
R_6	1	Power = 125 mw	0.45	0.00362	0.45
R_7	1	Power = 125 mw	0.101	0.00081	0.101
R_8, R_9	2	Power = 125 mw	0.114	0.00091	0.114
R_{10}	1	Power = 125 mw	6.24	0.05	6.24
R_{11}	1	Power = 125 mw	0.51	0.004	0.5
R_{12}	1	Power = 125 mw	0.142	0.00114	0.142
R_{13}	1	Power = 125 mw	13.2	0.105	13.2
Diodes 1N 3125 $CR_{1,2,3,4,5}$	5			0.029	
Capacitors $C_{1,2,3,4,5}$	5			0.6	

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Table 5-6. Probability - Stress Ratio (Continued)

SQUIB SIMULATOR					
				Max Rate at 80 C	
Part No. and Description	Quantity	Mfg Recommended Maximum Use	Actual Max Use (Milliwatts)	Ratio	Absolute (Milliwatts)
CR 6, 7 -C5B	2	100	40.0	0.40	40.0
CR4 - 1N941B	1	1000	225	0.225	225
CR1, 2, 3, 5-1N3547	4	250	90.0	0.360	90.0
CB1-CRK. & BRKR.	1				
K_1 - Relay	1	7000	2430	0.347	2430
R_1	1	Power = 5000	3000	0.600	3000
R_2	1	Power = 500	24.2	0.0484	24.2
R_3	1	Power = 500	195	0.390	195
R_4	1	Power = 500	0		
R_5	1	Power = 500	10.7	0.0214	10.7
R_6	1	Power = 500	336	0.672	336
R_7	1	Power = 10,000	3700	0.370	3700
R_8	1	Power = 500	29.2	0.0583	29.2
R_9	1	Power = 500	71.8	0.144	71.8
R_{10}	1	Power = 500	11.55	0.0231	11.6
C_1, C_2, C_3	3			0.060	

Table 5-6. Probability Analysis - Stress Ratio (Continued)

RESET SET TOGGLE FLIP FLOP T-613					
				Max Rate at 80 C	
Part No. and Description	Quantity	Mfg Recommended Maximum Use	Actual Max Use (Milliwatts)	Ratio	Absolute (Milliwatts)
Q_1, Q_2 2N428	2	Power = 150 mw BVCEO = -12 volts	44.1	0.294	44
Q_3, Q_4 2N1478	2	Power = 250 mw BVCEO = -20 volts	21.8	0.0873	21.8
CR_1 through 6 1N3125	6	Power = 80 mw BV = -20 volts	0.272	0.0034	0.27
R_1	1	Power = 125 mw	1.94	0.0155	1.9
R_2	1		1.94	0.0155	1.9
R_3	1		0		
R_4	1		37.9	0.303	38
R_5	1		2.74	0.0219	2.74
R_6	1		0.728	0.00583	0.73
R_7	1		98.0	0.784	98
R_8	1		0.0003	2×10^{-6}	0.0003
R_9	1		0.00404	3.23×10^{-5}	0.004
R_{10}	1		40.1	0.32	40
R_{11}	1		0		
R_{12}	1		3.16	0.0253	3.16
R_{13}	1		0.170	0.00136	0.17
R_{14}	1		0		
R_{15}	1		0		
R_{16}	1		0		
R_{17}	1		0.170	0.00136	0.17
R_{18}	1		15.9	0.127	16
R_{19}	1		0.866	0.00693	0.87
R_{20}	1	Power = 125 mw	0.334	0.00266	0.33
$C_{1, 2, 3, 4, 5, 6, 7}$	7				

Table 5-7. Probability Analysis - Failure Rate Percent/10^3 Hours

VOLTAGE COMPARATOR T-172					
Part No. and Description	Quantity	Failure Rate Generic (Chart)	K_a Application Factor Curves	Failure Rate Actual	Failure Rate Total
Q_1 2N 1478	1	0.090	0.96	0.0864	0.0864
Q_2 2N 1478	1	0.090	0.96	0.0864	0.0864
Q_3 2N 1304	1	0.090	0.96	0.0864	0.0864
Q_4 2N 1304	1	0.090	0.96	0.0864	0.0864
R_1	1	0.0004	0.2	0.0008	0.0008
R_2	1	0.0004	1.0	0.0004	0.0004
R_3	1	0.0004	0.8	0.00032	0.00032
R_4	1	0.0004	1.8	0.00072	0.00072
R_5	1	0.0004	0.5	0.00020	0.00020
R_6	1	0.0004	1.5	0.0006	0.0006
R_7	1	0.0004	0.6	0.00024	0.00024
R_8	1	0.0004	0.8	0.00032	0.00032
R_9	1	0.0004	1.7	0.00068	0.00068
R_{10}	1	0.0004	0.9	0.00036	0.00036
Connections, solder	32	0.0004	---	0.0004	0.0128
Connections, electrical	5	0.0035	---	0.0035	0.0175

K_a = Application Constant
R = Resistor
Q = Transistor
Total Failure Rate = 0.3804 Percent/10^3 per circuit per module

Table 5-7. Probability Analysis - Failure Rate Percent/10^3 Hours (Continued)

DUAL PULSE "AND" GATE T-448					
Part No. and Description	Quantity	Failure Rate Generic (Chart)	K_a Application Factor Curves	Failure Rate Actual	Failure Rate Total
Q_1 2N438	1	0.09	1.3	0.117	0.117
Q_2 2N428	1	0.09	1.3	0.117	0.117
CR_1 1N3125	1	0.02	0.8	0.01665	0.01665
R_1	1	0.0004	0.55	0.00022	0.00022
R_2	1	0.0004	2.0	0.0008	0.0008
R_3	1	0.0004	0.4	0.00016	0.00016
R_4	1	0.0004	1.0	0.0004	0.0004
R_9	1	0.0004	0.2	0.00008	0.00008
R_{11}	1	0.0004	0.625	0.00025	0.00025
R_{12}	1	0.0004	0.65	0.00026	0.00026
R_{13}	1	0.0004	1.8	0.00072	0.00072
R_{14}	1	0.0004	0.3	0.00012	0.00012
R_{15}	1	0.0004	0.6	0.00024	0.00024
C_1, C_3, C_5, C_6	4	0.005	0.06	0.0003	0.0003
Connections, Solder	32	0.0004	---	0.0004	0.0128
Connections, Electrical	4	0.0035	---	0.0035	0.0140
Total Failure Rate = 0.2810 Percent/10^3 hrs per circuit four circuits per module = 1.1240 percent/10^3 hrs					

Table 5-7. Probability Analysis - Failure Rate Percent/10^3 hrs (Continued)

ONE-SHOT MULTIVIBRATOR T-166					
Part No. and Description	Quantity	Failure Rate Generic (Chart)	K_a Application Factor Curves	Failure Rate Actual	Failure Rate Total
Q_1 2N428	2	0.084	1	0.084	0.168
Q_2 2N428	1	0.084	1	0.084	0.084
Capacitors $C_{1,2,3,4,5,}$	5	0.029	1	0.029	0.145
R_1	1	0.0297	1	0.0297	0.0297
R_2	1	0.0297	1	0.0297	0.0297
R_3	1	0.0297	1	0.0297	0.0297
R_4	1	0.0297	1	0.0297	0.0297
$R_5 R_{14}$	2	0.0297	1	0.0297	0.0594
R_6	1	0.00043	1	0.00043	0.00043
R_7	1	0.00043	1	0.00043	0.00043
$R_8 R_9$	2	0.00043	1	0.00043	0.00086
R_{10}	1	0.00043	1	0.00043	0.00043
R_{11}	1	0.00043	1	0.00043	0.00043
R_{12}	1	0.00043	1	0.00043	0.00043
R_{13}	1	0.00043	1	0.00043	0.00043
Diodes $CR_{1,2,3,4,5}$					
Total Failure Rate for four circuits/module = 2.36 percent/10^3 hrs x = 0.59064 percent/10^3 hrs per circuit MTBF = $\frac{1}{\text{Failure Rate}}$ = 42,500 hrs					

Table 5-7. Probability Analysis - Failure Rate Percent 10^3 hours (Continued)

SQUIB SIMULATOR					
Part No. and Description	Quantity	Failure Rate Generic (Chart)	K_a Application Factor Curves	Failure Rate Actual	Failure Rate Total
CR6, 7 - C5B	2	0.060	0.260	0.01560	0.31200
CR4 - 1N941B	1	0.015	0.180	0.00270	0.00270
CR1, 2, 3, 5-1N3547	4	0.020	0.210	0.00420	0.01680
CB1 -Circuit Breaker	1	0.0158	0.100	0.00158	0.00158
K1 - Relay	1	0.040	1.50	0.06000	0.06000
R_1	1	0.020	2.50	0.05000	0.05000
R_2	1	0.006	0.80	0.00480	0.00480
R_3	1	0.006	2.60	0.01560	0.01560
R_4	1	0.006	1.00	0.00600	0.00600
R_5	1	0.006	0.70	0.00420	0.00420
R_6	1	0.006	3.00	0.01800	0.01800
R_7	1	0.035	2.70	0.09450	0.09450
R_8	1	0.006	1.00	0.00600	0.00600
R_9	1	0.006	1.70	0.01020	0.01020
R_{10}	1	0.006	0.70	0.00420	0.00420
C1, C2, C3	3	0.005	0.06	0.00300	0.00300
Connections, Solder	17	0.0004		0.00040	0.00680
Connections, Electrical	11	0.0035		0.00350	0.03850
Total Failure Rate = 0.6549 percent/10^3 hr					

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Table 5-7. Probability Analysis - Failure Rate Percent/10^3 hours (Continued)

RESET SET TOGGLE FLIP FLOP T-643					
Part No. and Description	Quantity	Failure Rate Generic (Chart)	K_a Application Factor Curves	Failure Rate Actual	Failure Rate Total
Q_1, Q_2 2N428	2	0.09	2.5	0.225	0.225
Q_3, Q_4 2N1478	2	0.09	2.5	0.225	0.225
CR_1 through CR_6 1N3125	6	0.02	2.5	0.051	0.051
R_1	1	0.0004	0.8	0.00032	0.00032
R_2	1		0.8	0.00032	0.00032
R_3	1		0.2	0.00008	0.00008
R_4	1		2.2	0.00088	0.00088
R_5	1		0.8	0.00032	0.00032
R_6	1		0.6	0.00024	0.00024
R_7	1		3.0	0.00120	0.00120
R_8	1		0.2	0.00008	0.00004
R_9	1		0.2	0.00008	0.00004
R_{10}	1		2.2	0.00088	0.00088
R_{11}	1		0.2	0.00008	0.00008
R_{12}	1		0.8	0.00032	0.00032
R_{13}	1		0.8	0.00032	0.00032
R_{14}	1		0.2	0.00008	0.00008
R_{15}	1		0.2	0.00008	0.00008
R_{16}	1		0.2	0.00008	0.00008
R_{17}	1		0.8	0.00032	0.00032
R_{18}	1	0.0004	1.0	0.00040	0.00040

Table 5-7. Probability Analysis - Failure Rate Percent/10^3 hours (Continued)

RESET SET TOGGLE FLIP FLOP T-643 (Continued)					
Part No. and Description	Quantity	Failure Rate Generic (Chart)	K_a Application Factor Curves	Failure Rate Actual	Failure Rate Total
R_{19}	1	0.0004	0.6	0.00024	0.00024
R_{20}	1	0.0004	0.6	0.00024	0.00024
C1-C7	7	0.005	0.06	0.00030	0.00210
Connections, Solder	20	0.0004		0.00040	0.00800
Connections, Electrical	12	0.0035		0.00350	0.04200
Total Failure Rate = 0.5596 percent/10^3 hr per circuit					
Failure Rate for two circuits per module = 1.1192 percent/10^3 hr					

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Table 5-8. Reliability Prediction

VOLTAGE COMPARATOR T-172					
Part Name and Description	Quantity	Part No.	Mfg No.	** Failure Rate $\%/10^3$ Hr	Total Failure Rate $\%/10^3$ Hr
Trans. Q1, Q2	2	2N 1478	*	0.09	0.18
Trans. Q3, Q4	2	2N 1304	*	0.09	0.18
Resistor R1, R3	2	*	*	0.0043	0.0086
Resistor R2	1	*	*	0.0043	0.0043
Resistor R4	1	*	*	0.0043	0.0043
Resistor R5	1	*	*	0.0043	0.0043
Resistor R6	1	*	*	0.0043	0.0043
Resistor R7	1	*	*	0.0043	0.0043
Resistor R8	1	*	*	0.0043	0.0043
Resistor R9	1	*	*	0.0043	0.0043
Resistor R10	1	*	*	0.0043	0.0043
Connections, Solder	32	*	*	0.0004	0.0128
Connections, Plug	5	*	*	0.003	0.015
Total Failure Rate = 0.42222 $\%/10^3$ hr per circuit per module					

* Unknown
** General failure rates obtained from Martin's Handbook (compiled from actual military application)

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Table 5-8. Reliability Prediction (Continued)

DUAL PULSE "AND" GATE T-448					
Part Name and Description	Quantity	Part No.	Mfg No.	** Failure Rate $\%/10^3$ hr	Total Failure Rate $\%/10^3$ hr
Trans. Q1	1	2N428	*	0.09	0.09
Trans. Q2	1	2N438	*	0.09	0.09
Diode CR1	1	1N3125	*	0.03	0.03
Resistor R1	1	*	*	0.0043	0.0043
Resistor R2	1	*	*	0.0043	0.0043
Resistor R3	1	*	*	0.0043	0.0043
Resistor R4	1	*	*	0.0043	0.0043
Resistor R9	1	*	*	0.0043	0.0043
Resistor R11	1	*	*	0.0043	0.0043
Resistor R12	1	*	*	0.0043	0.0043
Resistor R13	1	*	*	0.0043	0.0043
Resistor R14	1	*	*	0.0043	0.0043
Resistor R15	1	*	*	0.0043	0.0043
Cap. C1, C3, C5, C6	4	*	*	0.0012	0.0048
Connections, Solder	32	*	*	0.0004	0.0128
Connections, Plug	4	*	*	0.003	0.010
Total Failure Rate = 0.2846 $\%/10^3$ hr per circuit four circuits per module = 1.1384 $\%/10^3$ hr					

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Table 5-8. Reliability Prediction (Continued)

RST FLIP FLOP T-643					
Part Name and Description	Quantity	Part No.	Mfg No.	** Failure Rate $\%/10^3$ hr	Total Failure Rate $\%/10^3$ hr
Trans. Q1, Q2	2	2N428	*	0.0900	0.1800
Trans. Q3, Q4	2	2N1478	*	0.0900	0.1800
Diodes CR1-6	6	1N3125	*	0.0300	0.1800
Resistor R1	1	*	*	0.0043	0.0043
Resistor R2	1	*	*	0.0043	0.0043
Resistor R3	1	*	*	0.0043	0.0043
Resistor R4	1	*	*	0.0043	0.0043
Resistor R5	1	*	*	0.0043	0.0043
Resistor R6	1	*	*	0.0043	0.0043
Resistor R7	1	*	*	0.0043	0.0043
Resistor R8	1	*	*	0.0043	0.0043
Resistor R9	1	*	*	0.0043	0.0043
Resistor R10	1	*	*	0.0043	0.0043
Resistor R11	1	*	*	0.0043	0.0043
Resistor R12	1	*	*	0.0043	0.0043
Resistor R13	1	*	*	0.0043	0.0043
Resistor R14	1	*	*	0.0043	0.0043
Resistor R15	1	*	*	0.0043	0.0043
Resistor R16	1	*	*	0.0043	0.0043
Resistor R17	1	*	*	0.0043	0.0043
Resistor R18	1	*	*	0.0043	0.0043
Resistor R19		*	*	0.0043	0.0043
Resistor R20	1	*	*	0.0043	0.0043
Caps. C1-C7	7	*	*	0.0012	0.0084

Table 5-8. Reliability Prediction (Continued)

RST FLIP FLOP T-64					
Part Name and Description	Quantity	Part No.	Mfg No.	** Failure Rate %/10^3 hr	Total Failure Rate %/10^3 hr
Connections, Solder	20	*	*	0.0004	0.0080
Connections, Electrical	12	*	*	0.0030	0.0360
Total Failure Rate = 0.6784 %/10^3 hr per circuit Failure Rate for two circuits/module = 1.3568 %/1000 hr					
SQUIB SIMULATOR					
Part Name and Description	Quantity	Part No.	Mfg No.	** Failure Rate %/10^3 hr	Total Failure Rate %/10^3 hr
SCR - CR6, 7	2	C5B	*	0.0600	0.1200
Zener - CR4	1	1N941B	1M9, 1Z	0.0150	0.0150
Diode - CR1-5	4	1N3547	*	0.0200	0.0800
Ckt_1 Board-CB1	1	*	PAM12-HK	0.0140	0.0140
Relay - K1	1	*	BR7X-E8	0.0250	0.0250
Resistor R1	1	*	*	0.0043	0.0043
Resistor R2	1	*	*	0.0043	0.0043
Resistor R3	1	*	*	0.0043	0.0043
Resistor R4	1	*	*	0.0043	0.0043
Resistor R5	1	*	*	0.0043	0.0043
Resistor R6	1	*	*	0.0043	0.0043
Resistor R7	1	*	*	0.0043	0.0043

[redacted]

Table 5-8. Reliability Prediction (Continued)

SQUIB SIMULATOR					
Part Name and Description	Quantity	Part No.	Mfg No.	** Failure Rate $\%/10^3$ hr	Total Failure Rate $\%/10^3$ hr
Resistor R8	1	*	*	0.0043	0.0043
Resistor R9	1	*	*	0.0043	0.0043
Resistor R10	1	*	*	0.0043	0.0043
Caps. C1, 2, 3	3	*	*	0.0025	0.0075
Connections, solder	17	-	-	0.0004	0.0068
Connections, electrical	11	-	-	0.0030	0.0330
Total Failure Rate = 0.3443 $\%/10^3$ hr					
ONE-SHOT MULTIVIBRATOR T-166					
Part Name and Description	Quantity	Part No.	Mfg No.	** Failure Rate $\%/10^3$ hr	Total Failure Rate $\%/10^3$ hr
Trans. Q_1 Q_2,	2	2N428	*	0.09	0.18
Trans. Q_3	1	2N428	*	0.09	0.09
Diodes $CR_{1,2,3,4,5}$	5	IN3125	*	0.03	0.15
Resistor R_1	1	*	*	0.015	0.015
Resistor R_2	1	*	*	0.015	0.015
Resistor R_3	1	*	*	0.015	0.015
Resistor R_4 R_{14}	1	*	*	0.015	0.015

Table 5-8. Reliability Prediction (Continued)

ONE-SHOT MULTIVIBRATOR T-166					
Part Name and Description	Quantity	Part No.	Mfg No.	** Failure Rate $\%/10^3$ hr	Total Failure Rate $\%/10^3$ hr
Resistor R_5	2	*	*	0.015	0.030
Resistor R_6	1	*	*	0.0043	0.0043
Resistor R_7	1	*	*	0.0043	0.0043
Resistor R_8 R_9	2	*	*	0.0043	0.0086
Resistor R_{10}	1	*	*	0.0043	0.0043
Resistor R_{11}	1	*	*	0.0043	0.0043
Resistor R_{12}	1	*	*	0.0043	0.0043
Resistor R_{13}	1	*	*	0.0043	0.0043
Capacitors $C_{1,2,3,4}$, C_x	5	*	*	0.004	0.020
Total Failure Rate = four circuits/module = $4\lambda = 2.28\%/10^3$ hr $\lambda = 0.5664\%/10^3$ hr MTBF = $\frac{1}{4}\lambda = 43,800$ hr					

Conclusions and Recommendations

The results of the Mandex worst-case d-c analysis reveal that the initiator subcircuits should operate reliably within worst-case allowable output extremes and should not fail due to worst-case variations in component part parameters and power stresses. The reliable circuit operation predicted by the computer analysis can be substantiated by the generic parts failure rate analysis conducted on the subcircuits. The AVCO probability of failure analysis yielded failure rates of 5.055 percent per 1000 hours for the EBW simulator and 0.655 percent per 1000 hours for the squib simulator. This is comparable to 5.268 percent per 1000 hours for the EBW simulator and 0.344 percent per 1000 hours for the squib simulator as compiled from the Martin reliability parts analysis. The combined analyses yield an

average failure rate of 5.162 percent per 1000 hours and an MTBF of 19.380 hours for the EBW simulator; and for the squib simulator 0.500 percent per 1000 hours and 200,000 hours, respectively.

Over-all subcircuit design has been found to be quite reliable. Wherever possible, reduction of ground and power-line noise has been attempted, utilizing bypass capacitors and multistep divider networks. An attempt has also been made to eliminate subcircuit interaction and the interaction between the system and GSE by means of blocking and suppression diodes. The input to the EBW comparator subcircuit has been protected against the inherent noisiness of the firing unit gap switch and variations in the self-test GSE by means of a clipping zener diode.

Due to the high operational reliability predicted by the preceding analysis, no design changes are recommended for the pyrotechnic initiator simulator units at this time. A failure mode and effects analysis shall be conducted and presented in a subsequent report.

It has been the purpose of this report to evaluate the performance of the pyrotechnic initiator simulator set (Model A14-003) by assessing its degree of reliable operation. This has been accomplished by analyzing its systems, namely the EBW and squib initiator simulators, and predicting failure rates and worst-case operational modes for the simulator subcircuits. The failure rate of the pyrotechnic initiator simulator set is equal to the summation of the failure rates of the total number of the EBW and squib simulators composing the set (see Table 5-9). This number will tend to vary, depending on the particular boiler plate involved. For example, the set used in conjunction with Boilerplate 6 contains six EBW and twenty-two squib simulators, while that contained in the Boilerplate 12 set are six and twenty-eight, respectively. Because of this variation, the analysis has not been predicated upon the end item but upon the two types of initiator simulators, so that the total system failure rate must be compiled by a summation as stated above for the particular pyrotechnic initiator simulator set of interest.

GSE TEST ACCOMPLISHMENTS

Qualification Test Plan

Reliability test activities during the fifth quarter centered around reviewing and amending design specifications, preliminary specifications, model specifications, or procurement specifications. Studies were made defining the qualification test plans for various subcontractors. In general, qualification tests on various parts, components, subassemblies, and higher levels of assembly shall be performed in accordance with the provisions of NCP-200-2, Paragraph 4.3 for major subcontractors.

Table 5-9. Boilerplate 6 Configuration Results

Initiator Simulator	No. Per System	Failure Rate %/1000 hr	Total Failure Rate %/1000 hr
EBW	6	5.162	30.972
SQUIB	22	0.500	11.000
		Total Failure Rate =	41.972
MTBF = 1/Failure Rate = 1/41.972 %/1000 hr MTBF = 2400 hr			

General Test Program Status

Though in the planning stages, a minimum test program is planned for GSE. The items, for test purposes, have been separated into the classes based on the contribution to the mission; the mission essential and mission nonessential items are treated about the same as the less critical items in the spacecraft. The nonessential items are given conventional qualification test only.

PLANNED ACTIVITIES

Analysis Group

1. Perform preliminary and final design analysis on PACE equipment and servicing equipment units that are categorized as mission essential equipment. Studies to include reliability predictions; stress analysis (Mandex d-c computer aided); failure mode and effects; participate in design reviews; and review all procurement specifications, specification control drawings, and design control documents and supply reliability inputs to these documents.

2. Develop additional computer capabilities to include mechanical stress studies and simplify existing Mandex computer programs to eliminate time consuming requirement for writing matrix equations.

3. Develop operational readiness program

Establish logic block diagram

Apportion MTBF's to units comprising logic diagrams

Establish and utilize communication lines between Reliability, Apollo Test Operations, Systems Integrations, Life Sciences, and Logistics to supply inputs and request pertinent information to fulfill the program objectives

Modify the existing Monte Carlo computer program as required to fit the needs of this program

Test Group

The sixth quarter of 1963 planned activity in the Ground Support Equipment area will continue with reviews and comments on procurement specifications. In addition, test procedures from major subcontractors and suppliers will be reviewed for approval and actual testing will be monitored at various test facilities. It is anticipated that the majority of testing in the first quarter will be accomplished in local areas, i. e., Culver City and Los Angeles.

VI. SUPPORT OPERATIONS

DATA MANAGEMENT

NONCONFORMANCE REPORTING SYSTEM

Phase I programming of the nonconformance reporting system was completed during the report period, and implementation of the system has progressed on schedule. Early de-bugging and coordination with format-initiating personnel has resulted in improved reporting and satisfactory input of raw data. Current effort is being directed toward accomplishing a more expeditious flow of data inputs. Data will be collated and distributed periodically as follows:

1. Nonconformance data – initial reports only (daily)
2. Nonconformance data – updated reports (weekly)
3. Nonconformance system problem summary (weekly)
4. Nonconformance failure mode – six months summary (weekly)
5. Nonconformance data – selected GSE report (monthly)
6. Nonconformance human factor summary (monthly)
7. Nonconformance material review summary (weekly)

Approximately 800 nonconformance items were reported during this quarter. Manufacturing discrepancies such as misdrilled holes, defective welds, adhesive deficiencies, dimensional errors, workmanship errors, etc., constituted the major portion of rejections. Twenty-five corrective NCAR action requests were initiated during the report period, three requiring engineering action, the remainder requiring manufacturing action. Thirty corrective NCAR actions were completed, some of which were initiated in the previous quarter period.

SUBCONTRACTOR DATA ACTIVITIES

Functional problems of equipment items reported are as follows:

1. Seventeen component calibration problems involving NASA-furnished equipment. The components were either returned to MSC or were accepted "as is."

2. Forty-three development-type problems on earth landing system components and 31 problem analyses were received from Northrop-Ventura Corporation. The problems involve stitching, material tears, broken vent lines, material burns, and reefing line cutter failures; the analyses resulted in redesign and revised rigging procedures.

3. Five development-type problems from Collins Radio Corporation involved Traveller Wave Guide Tubes manufactured by Hughes Electronics. The tubes were returned to Hughes for analysis.

Meetings were held with Lockheed Propulsion, Pratt & Whitney, AiResearch, Rocketdyne, Northrop-Ventura, and Aerojet to define data requirements and reporting formats as delineated in Procurement Specification MC 999-0025.

QUALIFICATIONS STATUS LIST

An updated qualifications status list was issued giving S&ID and suppliers reported status of 466 components used in 11 major systems, including G.S.E.

TOTAL RECORDS ACCESS CONTROL (TRAC) SYSTEM

The parts library tape, listing part numbers and a descriptor, was completed on schedule. At present, there are approximately 10,000 parts on the tape; additional parts will be included as engineering drawings are released. Future inputs to the parts library tape will include NASA traceability codes and material classification (generic) codes.

INTERSERVICE DATA EXCHANGE PROGRAM (IDEP)

Approximately 700 IDEP reports were distributed during the quarter; ten reports were contributed to the program plan. An audio/visual presentation outlining the purpose, function, and benefits of IDEP information was given to engineering, manufacturing, and development laboratory personnel. A copy was also given to the NASA Apollo Reliability Manager for NASA use.

PERFORMANCE ANALYSIS AND TEST HISTORIES (PATH) SYSTEM

Basic system definitions were completed for the PATH Phase I system, providing for storage, reporting, and analyses of test results and

specifications. Model test data samples were compiled, and sample outputs have been developed from them. An identification scheme for test environments also is being developed. PATH requirements are being appraised with respect to machine processing implementation, and further coordination and resolution of definition will be carried on during this phase of system development.

OPERATING TIME AND CYCLES DATA SYSTEM

A proposed procedure was completed for the acquisition and handling of operating time on time-sensitive equipment. Data processing printouts were designed to meet quality assurance requirements. Figure 6-1 illustrates the flow of operating time and cycle data.

The system being developed will be able to support the reliability program by providing the following data:

Accumulative time and cycles
Accumulative cycles
Mean-time-between-failures
Mean-time-between-removals
Failure rates
Removal rates
Limited life
Remaining life

In addition, this data will support the following computer programs:

Population density function
Cumulative distribution function
Reliability values
Spares requirements
Weibull hazard rate
Normal hazard rate
Gamma hazard rate
Beta hazard rate

IDENTIFICATION AND TRACEABILITY CONFIGURATION SYSTEM

Programming ground rules and number and sequence of programs involved are being determined. A Data transmittal sheet and automated output formats were drafted and submitted for review. Implementation of the program is estimated to occur in eight months.

END-ITEM DATA PACKAGE

Quality assurance computer systems were reviewed to determine their optimum use in preparation of the data package. The following data and report formats were defined as computer output requirements for inclusion in the data package:

1. Failure and corrective action summaries containing all failures experienced during assembly and final acceptance tests.

2. Material review summaries identifying end-item articles that have required material review action and indicating review disposition in each instance.

3. Operating time and cycles reports summarizing the accumulated time and cycles on all time-sensitive parts.

4. Final configuration reports listing all deliverable end-item parts subject to configuration accounting.

5. Replacement data report summarizing all part replacements made during installation, test, and final checkout operations conducted by S&ID.

6. Systems acceptance test data reports showing all operational parametric data recorded during installation operations and final acceptance test of the end-item, and all major systems incorporated.

MASTER CODE TABLES

A master code table document (S&ID 62-1412) was issued containing codes to be used by data management personnel in the transmittal of all types of data to Data Processing. Revisions and additions to the codes will be made as the project develops.

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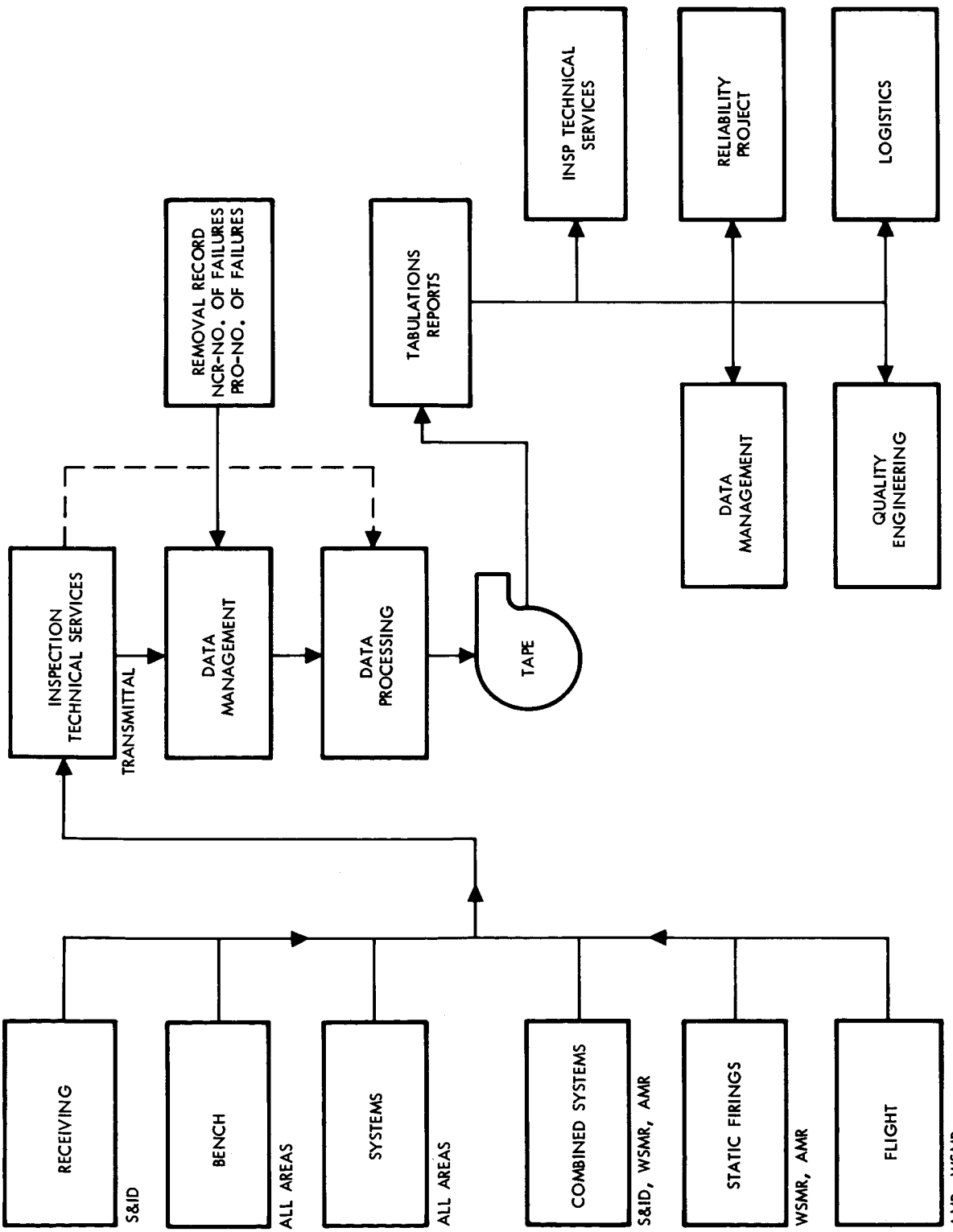

Figure 6-1. Flow of Operating Time/Cycle Data

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TRAINING AND EDUCATION

Educational courses and briefings presented during this report period are shown in Table 6-1.

Table 6-1. Reliability and Briefing Courses

Course Title	Number of Classes Given	Average Attendance	Total Hours
Fundamentals of Reliability Mathematics	11	11.5	180
Design Analysis Techniques	10	10.5	158
Symbolic Logic	6	12	108
Reliability Program Plan	1	56	140
Interservice Data Exchange Program	4	18	36
Supplier Briefings	1	38	38
Total	33		660

A list of educational publications including those that were completed and submitted for approval during this report period is shown in Table 6-2.

The Apollo Reliability Program Plan, high reliability parts, and design review indoctrination scripts were completed for closed-circuit television production. A series of 20-minute motivational scripts were also completed during the report period. Subject matter in these scripts include: Why space, 100,000 astronauts, logistic role and product assurance, and life support in space.

SUPPLIER BRIEFING

A symposium was held for potential suppliers in February 1963. Approximately 32 representatives of 16 companies associated primarily with valves and similar hardware attended. Briefings were conducted concerning

Table 6-2. Reliability Education Publications

Title	Pages	Status
Reliability Techniques	36	Published June 1962
Reliability Definitions	113	Published August 1962
Fundamentals of Reliability Mathematics	94	Published November 1962
Apollo Reliability Control Program	28	Published December 1962
Symbolic/Boolean Logic	23	Published February 1963
Confidence in Reliability	35*	Submitted for approval
Design Analysis Techniques, Volume I	80	Published February 1963
Design Analysis Techniques, Volume II	89	Published February 1963
Design Analysis Techniques, Volume III	95*	Submitted for publication
Computer Methods of Analysis, Volume IV	100*	Submitted for publication
Advanced Reliability Mathematics	60*	50 percent complete
Reliability Apportionment	30*	30 percent complete
Design of Experiments	15*	5 percent complete

*Estimated

contracts, pricing, engineering, and reliability. Specific reliability areas discussed were as follows:

1. IDEP material pertaining to specific subject to non-members on a need-to-know basis

2. Definitions and desired scope of various development and qualification test programs

3. Reliability management requirements in relatively small companies and definitions of what constitutes a minimum reliability program emphasizing that, as a minimum, an individual (or individuals) should be assigned the responsibility for a reliability program and should be provided with the authority to discharge the responsibility.

Each supplier representative completed a critique sheet at the conclusion of the meeting. The concensus of the remarks indicated that the meeting was profitable. Future symposiums are being considered for supplier education.

PLANNED ACTIVITIES

Planned activities which are scheduled for completion during the next quarter are as follows:

1. Approximately 1800 student-hours of instructions for reliability educational courses will be presented.

2. Material for courses will be prepared in advanced reliability mathematics and prediction and assessment.

3. Supplier educational program plan will be completed and presented for approval.

4. The first closed-circuit television will be presented to Apollo personnel during the next quarter.

COMPONENT TECHNOLOGY

RELIABILITY PARTS MANUAL

Volume II of the Reliability Parts Manual (Preferred Parts) was completed and distributed to all pertinent areas. This manual provides a selection of preferred parts estimated to fulfill approximately 80 percent of Apollo electronic, electrical, and electro-mechanical piece-parts requirements. It is intended to be utilized as a guide for the designer in the selection of high-reliability parts. In addition, it should serve as an aid to the reliability analyst in the evaluation of piece-parts utilized in the Apollo systems. Standardization will be enhanced while reliability levels will be increased.

A unique aspect of this document is the listing of both high-reliability and general-use items which in all cases are electrically identical. Moderate-cost items may, therefore, be used for design evaluation mock-ups and in noncritical areas. Furthermore, the manual provides a rapid, economical, and effective means of up-grading circuitry without extensive redesign when necessary.

To ensure against accidental degradation of parts through misapplication, approximately 500 pages of part data are provided, including approved maximum usage ratings and other pertinent application information.

Exhibited part reliability history was a major consideration in the selection of parts. However, proper documentation also is an important factor. Because required control documents did not exist, Component Technology generated the specification control drawings (SCD's) and procurement specifications for the items contained in this manual.

PRODUCT AND SPECIFICATION CONTROL DRAWINGS

Seventeen major product specifications and 136 SCD's will be required to provide proper specifications for the parts listed in the Reliability Parts Manual. Eight of the product specifications have received final approval and have been released; eight are in the final-approval-coordination stage and one is in final draft. All specifications will be released during the next quarter.

Twenty-three SCD's have been distributed to proposed vendors for their evaluation. The remaining 113 SCD's are in various stages of completion. Early May 1963 is the estimated completion date for all SCD's.

[redacted]

Nineteen procurement specifications and four minor-usage-part SCD's were reviewed by Component Technology at the request of Apollo Reliability. Specific deficiencies were noted and corrected to up-grade the documents to required reliability levels.

A review of 31 parts utilized in the ITT system was completed, with a recommendation that the RS-20 resistor be replaced by a Minuteman type.

SUPPLIER SURVEILLANCE

During the past quarter, a total of 35 preaward surveys were conducted including two re-surveys. Companies surveyed, their location, and the Apollo product are summarized in Table 6-3. Total surveys conducted to date is 153.

An updated summary of all surveys conducted during 1962 was released and distributed to Engineering, Purchasing, and Quality Control. This supplier status summary will be amended to include recent surveys during the next quarter.

Preparation of a schedule of post-award surveys is in progress and will be finalized in May. Continuous supplier surveillance will be accomplished prior to and concurrent with hardware deliveries.

Table 6-3. Supplier Surveys

Equipment	Name of Supplier	Location
Mission simulator	ACF Industries Curtis-Wright General Precision Goodyear Aircraft	Riverdale, Md. E. Paterson, N.J. Binghamton, N.Y. Akron, Ohio
Ground cooling cart	Air-A-Plane Advanced Structures Garrett Ltd. Hamilton Standard	Norfolk, Va. Lynwood, Calif. Ontario, Canada Hartford, Conn.
KMC High-gain antenna (2) Radome	Aero-Geo-Astro Raytheon General Dynamics (Convair-Pomona) Corning Glass H.I. Thompson	Alexandria, Va. Bedford, Mass. Pomona, Calif. Corning, N.Y. Gardena, Calif.
Oxygen, fuel and helium tanks	Ling-Temco-Vought Marquardt	Dallas, Tex. Ogden, Utah

Table 6-3. Supplier Surveys (Continued)

Equipment	Name of Supplier	Location
Helium tanks	Aerojet-General Airtek Menasco (resurvey)	Downey, Calif. Compton, Calif. Burbank, Calif.
Waste management blower	Globe Industries Ratron Western Gear	Dayton, Ohio Woodstock, N. Y. Pasadena, Calif.
Operational beacon antenna	Emertron Ryan Electronics	Silver Springs, Md. San Diego, Calif.
Explosive bolts	Hi-Shear Space Ordnance Systems Talley Industries	Torrance, Calif. Compton, Calif. Mesa, Arizona
Heat exchanger	Benson Mfg. Filtra-Sonic Houston Fearless Stewart Warner	Kansas City, Mo. Sun Valley, Calif. Torrance, Calif. Indianapolis, Ind.
Dynamic motion simulator	J. W. Fecker Shawnee Industries	Pittsburgh, Pa. Shawnee, Okla.
Instrumentation package	Fairchild Controls	Hicksville, N. Y.
Pressure and temperature transducers	Micro-Systems	San Gabriel, Calif.
Helium regulators	Fairchild Stratos (resurvey)	Manhattan Beach, Calif.

APPENDIX A
TECHNICAL PROPOSALS REVIEWED

Equipment	Company	Location
Pulse Code Modulation Carry-On, Model C14-210	Applied Electronics Corp.	Metuchen, N. Y.
Apollo Mission Simulator	General Precision (Link Div)	Binghampton, N. Y.
	Melpar, Inc. (Sub of Westinghouse)	Falls Church, Va.
	Minneapolis-Honeywell (Reg. Co., Ord Div)	Duarte, Cal.
	Goodyear Aircraft (Avionics Div)	Akron, Ohio
	McDonnell Aircraft (Electronics)	St Louis, Mo.
	American Car and Foundry Industries	Riverdale, Md.
	Curtis-Wright Corp. (Electronic Div)	East Patterson, N. J.
Helium Disconnects	Fairchild-Stratos (Western Branch)	Manhattan Beach, Cal.
	On-Mark Coupling, Inc.	Los Angeles, Cal.
	B. H. Hadley	Pomona, Cal.
	Schultz Tool and Mfg Co.	San Gabriel, Cal.
Propellant Utilization System	Acoustica Assoc (Advance System Div)	Inglewood, Cal.
	Avien, Inc.	Woodside, N. Y.
	Liquidometer Corp.	Long Island, N. Y.
	Minneapolis-Honeywell (Aeronautical Div)	Minneapolis, Minn.
	Simmonds Precision Products, Inc.	Vergennes, Vt.
	Space Technology Laboratory	Redondo Beach, Cal.
	Trans-Sonics, Inc.	Burlington, Mass.

Equipment	Company	Location
Propellant Disconnects	Schultz Tool & Mfg Co. Fairchild-Stratos (Western Branch) On-Mark Coupling, Inc. J. C. Carter Co. Snap-Tite, Inc. B. H. Hadley Cal Mec Manufacturing Corp.	San Gabriel, Cal. Manhattan Beach, Cal. Los Angeles, Cal. Costa Mesa, Cal. Union City, Pa. Pomona, Cal. Los Angeles, Cal.
SPS H_e Solenoid	Futurecraft Corp. Consolidated Controls Corp. Vacco Valve Co. APCO Accessory Products Co.	City of Industry, Cal. Inglewood, Cal. El Monte, Cal. Wilmington, Mass.
Bulkhead Electrical Feedthru for Apollo Command Module (Connector)	Amphenol-Borg. Electrical Corp Amphenol Western Connector Div. Cannon Electric Co Cole Electric Co Physical Sciences	Chatsworth, Cal. Phoenix, Ariz. Culver City, Cal. Pasadena, Cal.
Electrical Power Transfer Switch	Kinetics Corp Filters, Inc. (Div of Gen. Batteries & Ceramics) Westinghouse Electric Corp. (Aero Elec) Planautics Corp.	Solano Beach, Cal. E. North Post, N. Y. Lima, Ohio Encenitas, Cal.
Flood Lighting	Grimes Mfg Co.	Urbana, Ohio

APPENDIX B

PROCUREMENT DOCUMENTS CONTRIBUTIONS

No.	Title
DCDC-056A	Design Coordination Document for Calibration Unit (Model C14-056) Apollo Part Task Trainer Connector, Flexible, SPS Feed Lines Deodorizer Pad, Crewman, Personal Statement of Work, Food and Food Packaging Requirements Ingestible Dentrifrice, Crewman, Personal Light, Portable, Crewman Module Leak Test Unit (Model S14-001) Spacecraft Vertical Transport Vehicle (Model H14-036) Temperature Measuring System Transfer Switches, Motor Operated, RCS Valve, Oxygen, High Pressure, Charging
MC 111-0001	Bolt, Explosive, Electrically Initiated
MC 144-0023A	Coupling, MMH & Anhydrazine System, 1/4 Inch Test Point Disconnect
MC 148-0006	Engine Sling, S/M Propulsion Engine
MC 251-0005A	Burst Disc Assembly, Monomethyl Hydrazine
MC 271-0019	Hose Assembly, Flexible Metal, C/M, General Purpose
MC 273-0009A	Coupling, Helium Tank, Fill Disconnect
MC 273-0010A	Coupling, Helium Tank, 1/4 Inch Fill Disconnect
MC 273-0011A	Coupling, Nitrogen Tetroxide Tank, 1/4 Inch Fill Vent Disconnect
MC 273-0012A	Coupling, Hydrazine/UDMH Tank, 1 Inch Fill Vent Disconnect

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No.	Title
MC 273-0018A	Coupling, Nitrogen Tetroxide Tank, 1-1/2 Inch Fill & Drain Disconnect
MC 273-0019A	Coupling, Nitrogen Tetroxide Tank, 1/4 Inch Fill & Drain Disconnect
MC 273-0020A	Coupling, Hydrazine/UDMH Tank, 1-1/4 Inch Fill & Drain Disconnect
MC 273-0021A	Coupling, Hydrazine/UDMH Tank, 1/4 Inch Fill & Drain Disconnect
MC 273-0022A	Coupling, Nitrogen Tetroxide Tank, 1 Inch Fill Vent Disconnect
MC 273-0024A	Coupling, Hydrazine/UDMH Tank, 1/4 Inch Fill Vent Disconnect
MC 282-0002A	Pressure Vessel, Helium
MC 282-0003A	Pressure Vessel, Helium
MC 282-0004A	Tank, Nitrogen Tetroxide
MC 282-0006A	Tank, Nitrogen Tetroxide
MC 282-0007A	Tank, Hydrazine/UDMH
MC 282-0008A	Tank, Hydrazine/UDMH
MC 284-0001A	Valve, Solenoid, Shut-Off, Helium
MC 284-0013A	Valve, Nitrogen, Tetroxide, Shut-Off, Solenoid
MC 284-0018A	Valve, Solenoid, Shut-Off, Helium
MC 284-0019A	Valve, Explosive Operated, NC
MC 284-0020A	Regulator Unit, Pressure, Helium
MC 284-0021A	Regulator Unit, Pressure, Helium
MC 284-0022A	Regulator Unit, Pressure, Helium

No.	Title
MC 449-0015A	Mass Flow Transducer
MC 449-0020	Linear Acceleration Transducer
MC 449-0021	Temperature Transducers
MC 449-0024	Fuel Cell Powerplant Mass Flow Transducer
MC 452-0038A	Power Switch and Over Current Relay, Motor Operated
MC 453-0005A	Cartridge, Electrically Initiated
MC 453-0009	Initiator, Electrical, Hotwire
MC 453-0010	Pyrotechnic Electrical Circuit Interrupter
MC 453-0010	Cartridge Assembly, Electrically Iniated, Detonator, Hotwire
MC 454-003A	Circuit Breakers, Hermetically Sealed
MC 455-0014A	Control Relay, General Purpose
MC 456-0006A	Central Timing Equipment
MC 478-0020A	DSIF Power Amplifier
MC 478-0022A	HF Transceiver
MC 478-0026B	Unified S-Band Equipment
MC 481-0001B	R&D Telemetry Antenna Equipment
MC 481-0002A	Recovery Antenna System
MC 481-005A	Operational Beacon Antenna System
MC 481-0008	High Gain Antenna Equipment
MC 495-0001C	Inverter, Power Static
MC 901-0001C	Earth Landing System, Parachute Subsystem

No.	Title
MC 901-0057A	Fuel Cell Powerplant Test Stand
MC 901-0067B	Rocket Engine, Bi-Propellant, C/M, RCS
MC 901-0068A	Back-up Valve, Waste Management System
MC 901-0080	Acoustic Measurement System
MC 901-0081A	Signal Conditioning Equipment
MC 901-0092A	ECS, Water-Glycol Service Unit
MC 901-0093A	Console, Checkout, ECS
MC 901-0094A	ECS, Low Pressure Gaseous Test Stand
MC 901-0095A	ECS, High Pressure Gaseous Test Stand
MC 901-0096A	ECS, Pressure Distribution Test Set
MC 901-0097A	ECS, Test Set, Stimuli Generator, Suit Loop
MC 901-0098A	ECS, Water-Glycol Trim Control Set
MC 901-0099A	ECS, Electrical Test Stand
MC 901-0100A	ECS, Liquid Test Stand
MC 901-0107A	Bulkhead Feedthrough, C/M, Pressurized, Foward Bulkhead
MC 901-0108A	S/M Rocket Engine Motor Nozzle Closure
MC 901-0109A	S/M Rocket Engine Motor Leak Test Set
MC 901-0110A	S/M Rocket Engine Motor Nozzle Plug
MC 901-0111A	S/M Rocket Engine Motor Nozzle Extension
MC 901-0112	Vibration Measuring System
MC 901-0113	Leak Detection System

No.	Title
MC 284-0024A	Check Valve Assembly, Helium
MC 284-0025A	Check Valve Assembly
MC 284-0026A	Valve, Relief, Absolute Pressure, Helium
MC 284-0027A	Valve, Relief, Pressure
MC 284-0045A	Valve, Monomethyl Hydrazine, Solenoid Actuated
MC 284-0062A	Valve, Relief, Pressure, Helium, C/M, RCS
MC 284-0068A	Check Valve, Parallel, Helium
MC 336-0001	Camera, Motion Picture, 16 MM
MC 336-0002	Camera, Still, Portable
MC 364-0001C	Heat Shield Ablative Panels
MC 411-0104	Ribbon Cables, C/M & S/M, Bonded
MC 414-0061A	Electrical Connectors, Special Purpose
MC 414-0065	Connector, Umbilical, Electrical, C/M to S/M
MC 414-0067A	Umbilical, Electrical, Launch Escape Tower
MC 432-0040	In-Flight Manual Test Set
MC 434-0014A	Fixture, Flood Lighting, C/M Interior
MC 435-0013	Recorder, Data Storage
MC 443-0068	Resistors, Fixed, Wire Wound, Power Type, High Rel
MC 443-0073	Resistors, Fixed, Wirewound, Precision, High Rel
MC 449-0005	Pressure Transducer
MC 449-0014A	Displacement Transducer

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No.	Title
MC 901-0114	Stress Measuring System
MC 901-0115	Trainer, Mission Simulator
MC 901-0117	Bomb, Sojar, C/M
MC 901-0123	Helium Transfer Unit
MC 901-0126	Modification Kit, Telemetry Ground Station
MC 901-0127	Actuating System, Astro Sextant Door
MC 901-0128	Television B/M Equipment
MC 901-0135	S/M Propulsion Engine Nozzle Extension Closure
MC 999-0002B	Electromagnetic Interference Control
MC 999-0008A	Documentation Requirements for Project Apollo Suppliers (Components)
MC 999-0023A	Communications and Data Subsystems
MC 999-0027	Connectors, Electrical System, Apollo, General Spec for
MC 999-0028A	ECS, GSE, General Specification
MC 999-0034	ECS General Specification
MC 999-0035	Communications and Data Subsystem, GSE, General Specification
MC 434-0015	Light, Beacon, Flashing
MC 461-0002A	Charger, Battery, Silver Oxide Zinc
MC 481-0006A	C/M Radome
MC 461-0003B	Battery, Storage, Zinc Silver Oxide
MC 901-0008B	System, Propellant, Quantity Indicating and Mixture Control

No.	Title
MC 284-0050A	Ventilating Check Valve, Waste Management System
MC 901-0063B	In-Flight Test System
	Hose Assembly, Umbilical, Suit
	Cables, Electrical, Flat, Spacecraft, 600V
DCDC-009A	Design Coordination Document for Crew Systems C/O Group
DCDS-044	Vacuum Cleaner Service Set
MC-183-0016	S/M Propulsion Engine Assembly and Handling Fixture
	Feedthrough Connector Assembly, Coaxial, Pressurized, Bulkhead
	PCM Carry-on System
	Shaver, Crewman, Personal
MC 161-0012	Controls, Push-Pull, Rigid
MC 461-0007	Battery, Spacecraft Storage, Pyrotechnic, Silver Oxide Zinc
	S/M to GSE Umbilical Receptacle
SPECIFICATION CONTROL DRAWINGS (SCD's)	
ME 413-0007	Coaxial Isolator, Broadband
ME 454-0007	Circuit Breaker, 3 Pole, Thermal
ME 454-0008	Circuit Breaker, Single Pole, Thermal
ME 261-0008	Seal, S/M Fairing & C/M
ME 124-0002	Latch, Flush Mounted
ME 126-0001	Clip, Spring, Tension
ME 126-0002	Clip, Spring, Tension
ME 126-0003	Clip, Spring, Tension

No.	Title
ME 147-0001C	Gear Assembly, Speed Decreaser, 22:1 Reduction
ME 121-0024	Case, Instrument
ME 181-0025	Case, Instrument
ME 181-0026	Case, Instrument
ME 181-0027	Case, Instrument
ME 181-0028	Case, Instrument
ME 181-0029	Chassis
ME 273-0034	Quick Disconnect, High Pressure, Socket
ME 273-0035	Quick Disconnect, High Pressure, Nipple
ME 273-0036	Quick Disconnect, High Pressure, Cap
ME 273-0037	Quick Disconnect, High Pressure, Plug
ME 282-0018	Tank, Pressure, Coolant, Water-Glycol, 2300 cu.in.
ME 282-0019	Tank, Accumulator
ME 282-0020	Tank, Pressure, Coolant, Water-Glycol, 3940 cu.in.
ME 402-0006	RF Signal Generator
ME 402-0007	Pulse Generator
ME 415-0009	Socket, Relay, Microminiature, 10 Contact
ME 432-0042	Counter, Electronic, Digital
ME 432-0043	Meter, Power, 2.5 Watt
ME 432-0045	Meter, Frequency, Direct Reading
ME 435-0015	Recorder, Analog, 8 Channel
ME 454-0005	Circuit Breaker, Single Pole, Trip Free

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No.	Title
ME 454-0006	Circuit Breaker, Double Pole, Trip Free
ME 454-0007	Circuit Breaker, Three Pole, Trip Free
ME 455-0019	Relay, Armature, Magnetic Latching, Microminiature
ME 455-0037	Relay, Armature, DPDT, Heavy Duty
ME 464-0036	Power Supply 0-15VDC
ME 464-0037	Power Supply, 0-36VDC
ME 470-0006	Computer, Digital, Real Time

APPENDIX C

SUBCONTRACTOR DOCUMENTS REVIEWED

No.	Title
	AEROJET
3865-9A Amend 1	Quality Control Plan
3865-45	Simulated High Altitude Test Plan
3865-49	Simulated High Altitude Subscale Nozzle Evaluation Test Plan
3865-49 Amend 1	Simulated High Altitude Subscale Nozzle Evaluation Test Plan
3865-2-7	Monthly Weight and Balance Report
3865-31-13	Drawing List
3865-515	Ablative Materials, Evaluation Report
3865-30-3	Quarterly Reliability Status Report
3865-14-13	Quarterly Progress Report
3865-31-14	Drawing List
3865-2-8	Monthly Weight and Balance Report
3865-01-6	Monthly Progress Report
3865-01-7	Monthly Progress Report
	AIRESEARCH
SS-1003-R Rev 3	Reliability Program Plan
SS-1017-R (7)	Monthly Progress Report

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No.	Title
SS-1013-R (8)	Monthly Progress Report Monthly Progress Summary
SS-1013-R (9)	Monthly Progress Report
RAD-SR-62-202 Rev 8	Drawing List
RAD-SR-62-202 Rev 9	Drawing List
RAD-SR-63-4	Monthly Progress Report
RAD-SR-62-249	Monthly Weight and Balance Report
RAD-SR-62-202 Rev 10	Drawing List
RAD-SR-62-102 Rev 2	Pricing and Facilities Support Documentation
RAD-SR-63-N	Quarterly Reliability Status Report
RAD-SR-63-32	Monthly Progress Report
RAD-TM-63-5	Solution of Stefan's Problem for Decomposing Materials
RAD-SR-63-46	Monthly Progress Report
BEECH	
13707A	Reliability Program Plan
13703	Program Plan
13705-1	Development Test Plan
13705-2	Qualification Test Plan
13705-3	End Item Acceptance Test Plan
13709	Quality Control Plan

No.	Title
BR 13714 (Jan)	Monthly Progress Report
BR 13718	Monthly Weight and Balance Report
BR 13804	Failure Mode Analysis
13761	Maintenance Equipment Requirements Specification
13737 F	Drawings, Materials, Process and Procurement Specification List
13714 (Feb)	Monthly Progress Report
13714 (Mar)	Monthly Progress Report
BP 13912	Test Procedure for Proof and Burst Tests on the Titanium Airite Pressure Vessels
BP 13815	Spherical Calorimeter, Thermal Conductivity Evaluations
BP 13818	Vibration Test, 9 Inch Diameter Insulation Model
COLLINS	
AR 290-1	Test Plan
AR 118-3	Quality Control Plan
AR 111-3	Program Plan
AR 243-1	Process Control Procedures
AR 120-4	Quarterly Progress Report
AR 253-1	Quarterly Quality Control Summary
AR 101-9	Monthly Progress Report
AR 293-1	Application of Sampling Plan

No.	Title
AR 245-1	Acceptance Test Procedure for R&D Multiplexer 635X-1
AR 246-1	Acceptance Test Procedure for R&D VHF OMNI Antenna and Radome 437H-1
AR 101-10	Monthly Progress Report Number Code Key for Failure Reports Failure Analyses Failure Report
	LOCKHEED
588-M-6-3	End-Item Acceptance Test Plan
588-P-7	Monthly Progress Report
588-P-8	Monthly Progress Report
588-P-9	Monthly Progress Report
	MARQUARDT
A 1014-C	Support Plan
A 1033-3	End Item Acceptance Test Plan
A 1031	Quarterly Progress Report
A 1011-6	Monthly Progress Report
A 1011-7	Monthly Progress Report
	MELPAR
N 1004. 04 (Jan)	Monthly Progress Report
N 1004. 04 (Feb)	Monthly Progress Report
N 1004. 04 (Mar)	Monthly Progress Report

No.	Title
MINNEAPOLIS-HONEYWELL	
A62-750 B4 (3)	Design Specification, SCS, B/M Equipment
A62-751 E1 (3)	Reliability Program Plan
A62-751A (2)	Program Plan
A62-751 C (2)	Test Plan
A62-768 B (4)	Quality Control Plan
A63-751 A (2)	Program Plan
A62-777A16 (2)	Configuration Management Plan
A62-752 C (10)	Monthly Progress Report
A62-776A6 (5)	Cost Analysis of Special Tools and Test Equipment
A62-777A5 (3)	Attitude Control Evaluator, Detailed Description
A62-777A21 (1)	Augmented Continuous Control Evaluator, Detailed Description
A62-760B 36.1 (2)	Initial Failure Analysis for Apollo SCS
A62-760B .1 (1)	Launch Escape SCS Studies Analysis Results
A62-760A 13 (1)	Six-Degree-of-Freedom Analog Simulation Study
A62-752C (11)	Monthly Progress Report
A63-752 C (12)	Monthly Progress Report
A63-752 C (13)	Monthly Progress Report
NORTHROP	
2524B	Manufacturing Plan
2523D	Over-all Test Plan

No.	Title
2516B	Program Plan Failure Data Sheets
NVT/62-581	Monthly Progress Report
2600	Quarterly Progress Report Failure Analysis Sheets
NVC/62-3579/734	Drawing List
2620-19	Monthly Failure Summary
NVC/63-11/734	Drawing List Inspection Disposition Sheets
NVC/63-93/734	Quarterly Summary of Quality Control
NVC/63-140/734	Drawing List
2540	Quarterly Reliability Status Report
2620-42	Monthly Failure Summary
NVC/63-278/734	Drawing List
NVT/63-20 (7)	Monthly Progress Report
NVT/63-44 (8)	Monthly Progress Report
D5020	Development Test Report, Apollo Drogue Mortar Receiving Rejection Reports Failure Analysis Supplements
PRATT & WHITNEY	
PWA 2118	End-Item Test Plan
PWA 2117	Quality Control Plan
PWA 2149	End-Item Acceptance Test Plan
PWA 2147	Qualification Test Plan

No.	Title
PWA 2146	Development Test Plan
PWA 2117A	Quality Control Plan
PWA 2115A	Reliability Program Plan
PWA 2113A	Program Plan
PWA 24-12	Drawing List
PWA 20-5	Monthly Quality Report
PWA 24-13	Drawing List
PWA 2131	Quarterly Progress Report
PWA 24-14	Drawing List
PWA 2152	Quarterly Reliability Status
PWA 20-7	Monthly Quality Report
PWA 20-6	Quarterly Quality Control Summary
PWA 24-15	Drawing List
PWA 7	Monthly Progress Report
PWA 24-16	Drawing List
PWA 24-17	Drawing List
PWA 24-18	Drawing List
PWA 24-19	Drawing List
PWA 20-8	Monthly Quality Report
PWA 11	Monthly Weight and Balance Report
PWA 24-20	Drawing List

No.	Title
PWA 24-21	Drawing List
PWA 8	Monthly Progress Report
	ROCKETDYNE
62 RC 11880	Cost Proposal
SERR2394-006	Reliability Program Plan Cost Substantiation for C/M RCS Engines Milestone Charts
	THIOKOL
A 006B	Reliability Program Plan
A 009B	Quality Control Plan
A 205	Monthly Progress Report
A 712	Drawing List
A 713	Drawing List
A 303	Quarterly Progress Report
A 714	Drawing List
A 603	Quarterly Summary of Quality Control
A 503	Quarterly Reliability Status Report
A 206	Monthly Progress Report
	EAGLE PITCHER
	Qualification Test Plan
1	Monthly Progress Report
2	Monthly Progress Report

No.	Title
3	Monthly Progress Report
4	Monthly Progress Report
ITT	
1	Reliability Program Plan